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HIGH PERFORMANCE SINGLE ROTOR HELICOPTER STUDY

Task 9R38-13-014-01

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prepared by :

SIKORSKY AIRCRAFT
UNITED AIRCRAFT CORPORATION
Stratford, Connecticut

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SER-50174

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HEADQUARTERS
U. S. ARMY TRANSPORTATION RESEARCH COMMAND
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FOREWORD

The final report contained in the following pages presents the results of a high performance helicopter research program performed under U. S. Army contract by Sikorsky Aircraft Division of United Aircraft Corporation.

The U. S. Army through the efforts of the Transportation Research Command (USATRECOM) at Fort Eustis, Virginia, has a series of programs underway to determine the gains that can be obtained in the performance of helicopters, both single rotor and tandem, through utilization of the latest design techniques.

This report which is a final report presents the results of a research program which Sikorsky has conducted to determine what gains in performance can be obtained for the single rotor helicopter. In this study, Sikorsky has taken into account background material from other research efforts conducted by the Army, industry and other Government agencies, such as the National Aeronautics and Space Administration.

The data presented show the gains that can be obtained in range and/or endurance, speed and productivity. The various available components that could be utilized in fabrication of a high performance single rotor helicopter have also been presented.

The report has been reviewed in detail by USATRECOM and the findings and recommendations set forth in the report are concurred with. The results are quite encouraging in supporting the individual efforts that have indicated a high performance design is well within the realm of the state-of-the-art. It is hoped reports such as this one will arouse the interest of those concerned with tomorrow's helicopter designs as well as those concerned with improving the helicopter of today.

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TABLE OF CONTENTS

		Page
	FOREWORD	iii
	LIST OF ILLUSTRATIONS	vii
	LIST OF SYMBOLS	xi
I	SUMMARY	1
II	INTRODUCTION	9
III	DESIGN ANALYSIS	11
	A. SELECTION OF BASIC AIRFRAME	11
	1. Airframes Considered and Rejected	11
	2. General Features of New Airframe Based on S-61 Powerplant System	15
	B. ESTIMATION OF PARASITE DRAG	18
	1. Importance of Parasite Drag	18
	2. Rotor Head Fairing Test Results	18
	3. Estimate of Total Aircraft Parasite Drag	22
	C. PERFORMANCE CALCULATIONS	24
	1. Method of Calculation	24
	2. Selection of Rotor	26
	3. Calculated Performance Results	27
	D. BLADE STRESS AND VIBRATION STUDY	31
	1. Method of Calculation	31
	2. Correlation of Theory and Experiment	32
	3. Allowable Blade Stresses	33
	4. Selection of Rotor	33
	E. FLYING QUALITIES STUDY	37
	1. Introduction	37
	2. Background to Flying Qualities Study	37
	3. Refined Flying Qualities Design Analysis	34
	4. Calculated Flying Qualities	40

		Page
	F. AIRCRAFT DESIGN DETAILS	45
	G. TRANSMISSION AND ROTOR SYSTEM STUDY	48
	H. GROUP EMPTY WEIGHT STATEMENT	50
IV	SPECIFICATION AND PERFORMANCE SUMMARY	51
	A. DIMENSIONAL DATA	51
	B. WEIGHTS	52
	C. POWERPLANT	52
	D. PERFORMANCE	53
	E. PRODUCTIVITY STUDY	54
V	CONFIGURATION MODIFICATION FOR ADDITIONAL PERFORMANCE IMPROVEMENTS	55
	A. PURE HELICOPTER WITH INCREASED ROTOR SOLIDITY	55
	B. INCREASED ROTOR SOLIDITY PLUS A TURBOJET ENGINE FOR AUXILIARY LIFT AND PROPULSION	58
	C. HELICOPTER WITH WING	62
	D. HELICOPTER WITH WING AND TURBOJET ENGINES (JET COMPOUND)	66
	REFERENCES	70
	DISTRIBUTION	143

ILLUSTRATIONS

Figure		Page
1	Sikorsky High Performance Helicopter	xiv
2	Comparative Cruise Speeds	3
3	Ferry Range Capability	4
4	Productivity Comparison	5
5	Transport Mission Study	6
6	Helicopter International Speed Record	9
7	Aircraft Considered in Design Study	71
8	General Arrangement High Performance Helicopter	73
9	Dimensional Comparison High Performance Helicopter With S-61 (HSS-2)	75
10	Full Scale Rotor Head-Pylon Model	77
11	Results of Rotor Head Fairing-Pylon Model Test	78
12	Numerical Integration Rotor Performance Method	79
13	Engine Airflow External Forces	80
14-17	Calculated Forward Flight Performance	81-84
18	Calculated Hovering Performance	85
19	Hovering Ceilings	86
20	Optimum Flight Conditions for Maximum Range	87
21	Ferry Range Characteristics of HPH	88
22	Range Characteristics at 175 knots	89
23	Range Characteristics at 150 knots	90
24	Theoretical Representation of Flexable Blade	91

Figure		Page
25-26	Flatwise Bending Time Histories	92-93
27	Bending Moment Amplitudes	94
28	Blade Tip Deflection Due to Flatwise Bending	95
29	Allowable Spar Vibratory Stress	96
30	Effects of Blade Twist on Flatwise Bending	97
31	Effects of Twist on Stress Distribution	98
32	Additional Effects of Twist	99
33	Effect of Tip Speed on Vibratory Stress	100
34	Effects of Other Variables on Vibratory Stress	101
35	Vibratory Stress for Selected Rotor	102
36	Comparison of Flatwise and Edgewise Stresses	103
37	Time History of Symmetrical Pullout Maneuver	104
38	Variation of Propulsive Force with Angle of Attack	105
39	Effect of Cyclic Pitch on Rotor Resultant Force	106
40	Airspeed Response with Normal and Reversed Control	107
41	Helicopter Flight Simulator	108
42	Body Aerodynamic Characteristics Used in Simulator Study	109
43-44	Static Trim for High Performance Helicopter	110-111
45	Longitudinal Cyclic Pitch Position Stability	112
46	Longitudinal Control Response with Zero Flapping Hinge Offset	113
47-48	HPH Longitudinal Maneuvering Response	114-115

Figure		Page
49-50	HPH Longitudinal Dynamic Stability Characteristics	116-117
51-52	HPH Lateral Static Trim Characteristics	118-119
53	Variation of Thrust with Collective Pitch	120
54	Collective Pitch Required for Coordinated Turns	121
55	Ferry Tank (800 Gal.)	122
56	Payload-Range Characteristics of the High Performance Helicopter	123
57	Effect of Flight Speed on Block Speed	124
58	Productivity, turn Around Time 5 Minutes	125
59	Productivity, Turn Around Time 20 Minutes	126
60	Payload-Range Characteristics, Modification (a)	127
61	Productivity, Modification (a)	128
62	General Arrangement HPH with Jet	129
63	Payload-Range Characteristics, Modification (b)	131
64	Productivity, Modification (b)	132
65	General Arrangement High Performance Helicopter with Wing	133
66	Payload-Range Characteristics, Modification (c)	135
67	Productivity, Modification (c)	136
68	General Arrangement Jet Compound Helicopter	137
69	Payload-Range Characteristics, Modification (d)	139
70	Productivity, Modification (d)	140

Tables

		Page
1	Performance Comparison	2
2	Physical Characteristics Used in Simulator	141

SYMBOLS

b	number of blades per rotor
B_{1s}	longitudinal cyclic pitch, coefficient of first harmonic sine term in expression for blade pitch variation with azimuth
c	blade-section chord
C_{Qd}	drag torque coefficient for one blade, drag torque/$\pi R^2 \rho (\Omega R)^2 R$
D	drag
e	distance from flapping hinge to axis of rotation
f	parasite area, parasite drag/q_o
F	force
H	total pressure
L	lift
n	normal load factor
p	static pressure
Q	specific range, nautical miles per pound fuel
q	dynamic pressure
R	rotor radius
RPF	rotor propulsive force, component of rotor resultant force in flight direction
T	thrust
V	velocity
W	gross weight of helicopter
α	angle of attack

θ	jet exhaust angle
θ_o	theoretical blade pitch angle at center of rotation
$\theta_{.75R}$	collective pitch measured at 75 percent rotor radius
ρ	mass density of air
σ	rotor solidity, $bc/\pi R$
ψ	blade azimuth angle measured from downwind position in direction of rotation
Ω	rotor angular velocity

Subscripts

f	fuselage
j	jet
x	longitudinal
o	free stream

SIKORSKY

HIGH PERFORMANCE HELICOPTER

FIGURE 1

SECTION I: SUMMARY

A design study of a high performance single rotor helicopter was conducted by Sikorsky Aircraft under contract to the U.S. Army Transportation Research Command. An artist's drawing of this aircraft is shown in figure 1. This helicopter has outstanding performance capabilities compared to present day standards yet could be built and flown in the immediate future. It is based to a large extent on the dynamic components of an existing helicopter, the HSS-2 (Sikorsky S-61) which has set a world record of maximum speed without payload of 192.9 miles per hour (167.5 knots).

The basic high performance helicopter (HPH) will cruise at 175 knots with a payload of 2 tons. Maximum speed with this payload is 182 knots, and the aircraft is capable of a maximum speed without payload of 195 knots (224 mph). Minimum design ferry range capability was set at 1600 nautical miles, but a value in excess of 2000 nautical miles was achieved with a vertical take-off. Higher speed and payload capabilities were achieved with various modifications of the basic aircraft. The HPH is inherently stable in forward flight, meeting Military specification flying qualities requirements without artificial stabilization of any kind.

The high performance helicopter incorporates a number of basic advances made possible by recent Sikorsky research efforts. These advances include (1) the achievement of exceptionally low parasite drag made possible by the evolvement of a workable low-drag rotor head fairing concept; (2) rotor performance calculation techniques unrestricted with respect to advance ratio, Mach number effects, or blade stall effects; (3) a flexible blade analysis which includes the interaction between flatwise bending deflection and blade load distribution, permitting rational selection of optimum blade design for adequate fatigue life compatible with performance requirements; and (4) a flying qualities analysis based on real time analog computer solutions of the advanced rotor performance calculation procedure, permitting accurate and continuous simulated flight of the aircraft in all regimes with a human pilot at the controls.

PERFORMANCE RESULTS

Comparative performance data for two current helicopters and the high performance helicopter in the basic and modified configurations are shown in Table 1 and in bar chart form in figures 2-5 following:

TABLE 1

PERFORMANCE COMPARISONS

Configuration		Mission	Gross Weight Lb.	Payload Lb.	Range N.Miles	Speed Knots
Current Helicopters (Basic Airframe, no special equipment)	S-58	Typical	13,000	4,300	100	87
		Ferry	14,000	0	1,010	85
	S-61	Typical	17,300	6,600	100	130
		Ferry	19,000	0	1,290	100
Basic High Performance Helicopter (HPH) (5 blades 2 T-58-8 engines)		Max. Speed, No Payload	8,700	0	50	195
		Max. Speed with Payload	13,600	4,300	85	182
		Design High Speed Cruise	13,600	4,300	100	175
		Optimum Cruise	17,000	7,700	100	150
		Ferry	18,000	0	2,080	125
HPH Modifications (a) 6 blades		Max. Speed, No Payload	9,100	0	50	195
		High Speed Cruise	15,400	5,600	100	175
(b) 6 blades + JT-12 Turbojet		High Speed Cruise	16,600	4,150	100	200
(c) 5 blades + 110 ft.^{2} wing		High Speed Cruise	18,000	7,950	100	175
(d) Jet Compound Helicopter, 5 blades + 200 ft.^{2} wing + 2 JT-12 Turbojets		High Speed Cruise	17,600	4,350	100	250

All performance figures at sea level except for ferry mission.

A notable item in Table 1 is that a speed of 195 knots can be achieved with a pure helicopter configuration; a still higher speed could have been obtained except for the installed power limitation.

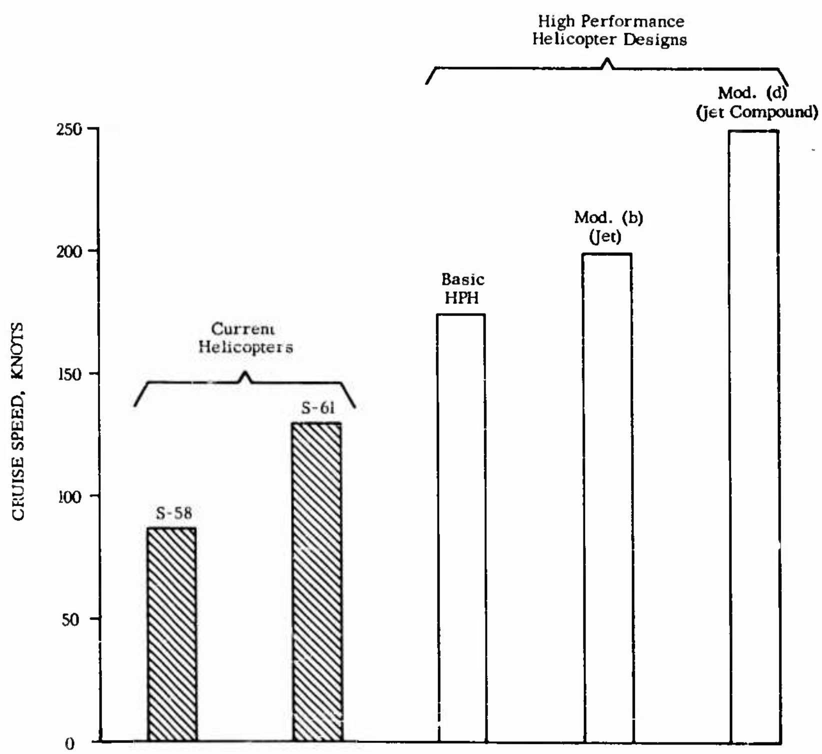

FIGURE 2. COMPARATIVE CRUISE SPEEDS

The cruising speeds (with payload) of the various configurations are shown in graphic form in figure 2. The high performance helicopter cruising at 175 knots provides nearly a 35 percent increase in cruise speed over the S-61 and a two-to-one increase in speed over the S-58 helicopter. Two of the HPH modifications cruise at still higher speeds; the 6-blade HPH plus turbojet (modification (b)) cruises at 200 knots, and the HPH plus wing plus two jets (jet compound, modification (d)) cruises at 250 knots.

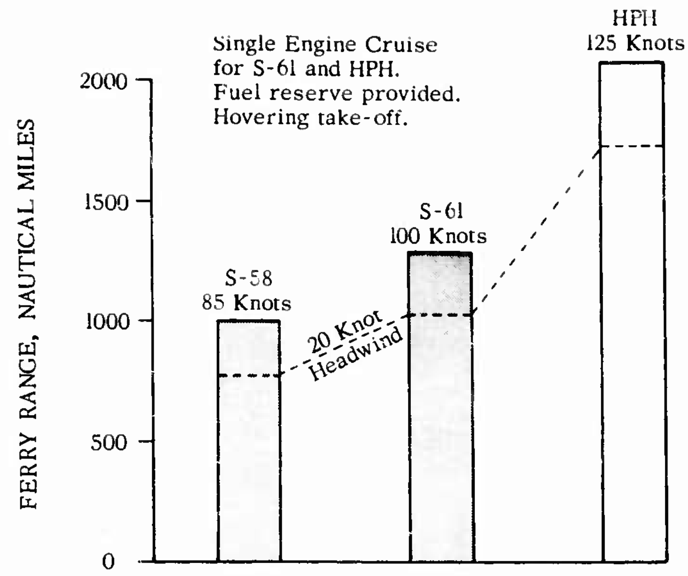

FIGURE 3. FERRY RANGE CAPABILITY

A comparison of ferry ranges is shown in figure 3. The HPH has a calculated still-air ferry range (vertical take-off) of 2080 nautical miles plus a one hour fuel reserve compared to approximately 1000 and 1300 nautical miles for the S-58 and S-61, respectively, for similar loading conditions. At a sacrifice in hovering performance, a ferry range in excess of 2300 nautical miles could be achieved with the HPH at a take-off gross weight of 20,000 pounds. The effect of a 20 knot headwind is also indicated in figure 3. While the reduction in miles is greater for the HPH than for the two current helicopters, the percentage range reduction is smaller because of the higher optimum cruise speed of the HPH. Even with the 20 knot headwind, the ferry range of the HPH exceeds the minimum target of 1600 nautical miles. Ferry range of the turbojet modifications of the HPH would be less than that shown because of the high fuel consumption rate of the jet, while range of the non-jet modifications would be approximately the same as for the basic HPH.

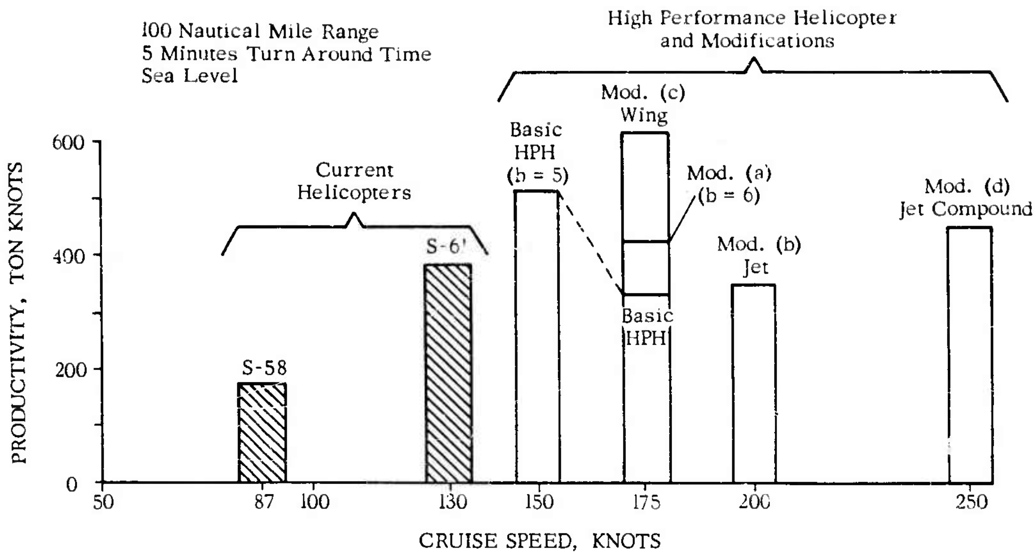

FIGURE 4. PRODUCTIVITY COMPARISON

Productivity (payload times block speed) for the various configurations is shown graphically in figure 4. This particular bar chart has a horizontal scale (cruise speed) as well as the vertical one. Productivity is a direct measure of the rate of accomplishing a useful transfer of material, and so is a vitally important parameter in any transport mission. It may be seen that the high performance helicopter and its modifications offer some very large gains over the current helicopters. The basic HPH has a considerably higher productivity at 150 knots than at the design cruise speed of 175 knots because of a considerably higher payload capacity at the reduced speed. An extra blade on the HPH (modification (a)) provides about 30 percent higher payload and productivity at the design speed, and a small (110 square feet) wing added to the basic HPH (modification (c)) adds about 85 percent at 175 knots. The payload and productivity of the HPH plus wing is limited only by the lifting capacity in hovering. Because of the high lift-drag ratio of the wing, the propulsive force demands on the rotor are not increased excessively, and the combination represents an extremely efficient method of increasing payload at the design speed. The productivity of this configuration was the highest of those considered in the study. It is believed that the benefits indicated for the wing are due to the combination of high flight speed and low parasite drag; the same benefits could not be obtained on less clean, slower helicopters. The high performance helicopter with small wing appears to offer the greatest potential of providing substantial increases in productivity with a minimum of complication.

The six blade HPH plus JT-12 turbojet (modification (b)) suffers a loss in productivity relative to the six-blade pure helicopter despite the increase in cruise speed to 200 knots. The reason for this is the rapid loss in rotor lifting capability with increasing speed in this speed regime, so that payload capacity is reduced. A better solution is to incorporate a lifting wing as well as auxiliary propulsion; the jet compound, which is the five-bladed HPH plus a 200 square foot wing plus two JT-12 turbojets (modification (d)), cruises at 250 knots with a better productivity than that of the helicopter plus jet only. Some price is paid for this speed, however, in that the productivity is less than that obtainable with other HPH configurations at lower speeds. It should be noted that this result might not apply if the thrust power available in forward flight were available to the main rotor in hovering.

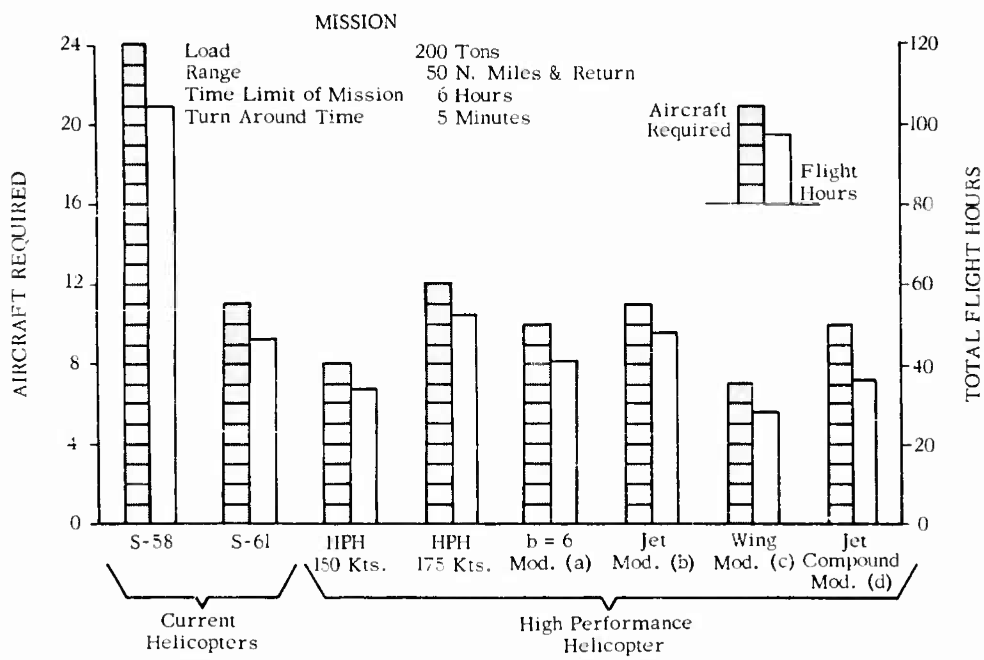

FIGURE 5. TRANSPORT MISSION STUDY

The advantages of the high productivity of the HPH for a particular transport mission is illustrated in figure 5, which shows the number of aircraft required and the total flight hours for a mission in which 200 tons of payload must be moved 50 nautical miles in a total of six hours. As indicated by the shaded bars, twenty-four S-58 helicopters or eleven S-61's would be required for this task, whereas eight HPH aircraft cruising at 150 knots or seven aircraft of the HPH plus wing configuration cruising at 175 knots would perform the same function. Thus fewer aircraft, and correspondingly fewer pilots, are required for a given transport operation because of the ability to carry a greater payload on a single flight and/or the ability to complete a greater number of round trips in a specified time interval.

Total aircraft flight time for the same mission, which is indicative of the maintenance time and manpower required to support the operation, is shown by the unshaded bars in figure 5. Total flight time follows the same pattern as the number of aircraft required, indicating that maintenance effort also derives substantial benefits from the high performance helicopter concept.

RECOMMENDATIONS

In view of the very substantial advantages afforded by the Sikorsky high performance helicopter over current types, it is recommended that the detail design of this helicopter proceed and a small number of aircraft be constructed for experimental investigation of high speed helicopter flight and a preliminary operational evaluation. The flight investigation should include correlation of performance, blade stresses, and flying qualities with predicted characteristics over the entire speed range, with emphasis on the higher speeds. It is also recommended that one of these aircraft be utilized as a research vehicle to explore the four modifications considered in this report. In particular, the high performance helicopter with wing should be investigated experimentally because of the very high productivity indicated for this configuration, and the jet compound configuration should also be investigated because of the very high speed potential of this rotary-wing type.

SECTION II: INTRODUCTION

Helicopters have been identified as machines capable of hovering essentially motionless in the air, or flying backwards or sideways as well as forwards. Because of this capability, the helicopter is particularly noted for the many rescue operations it has performed and other "skyhook" missions in which hovering plays a predominate role. Forward flight capability, on the other hand, has not been so notable because of the generally superior capabilities of fixed-wing aircraft in this flight regime. Despite a steady increase with time of the helicopter speed record (figure 6), a belief has arisen in some circles that the helicopter has nearly reached its ultimate development and is not capable of substantial further improvement. Various recent studies, such as those reported in References 1-3, take the opposite view that substantial performance improvements are possible; pure helicopters are believed capable of normal operational cruise speeds (as opposed to record speeds) on the order of 175 knots or more with ferry ranges on the order of at least 1600 nautical miles. Other forms of rotary wing aircraft, such as the compound helicopter, are believed capable of substantially higher speeds and ranges. Productivity, or payload times block speed, which is a measure of the rate of accomplishing a useful transfer of material, is believed to be subject to similar improvements, with a resultant decrease in the number of aircraft required to perform a given mission and corresponding reductions in pilots and maintenance personnel required.

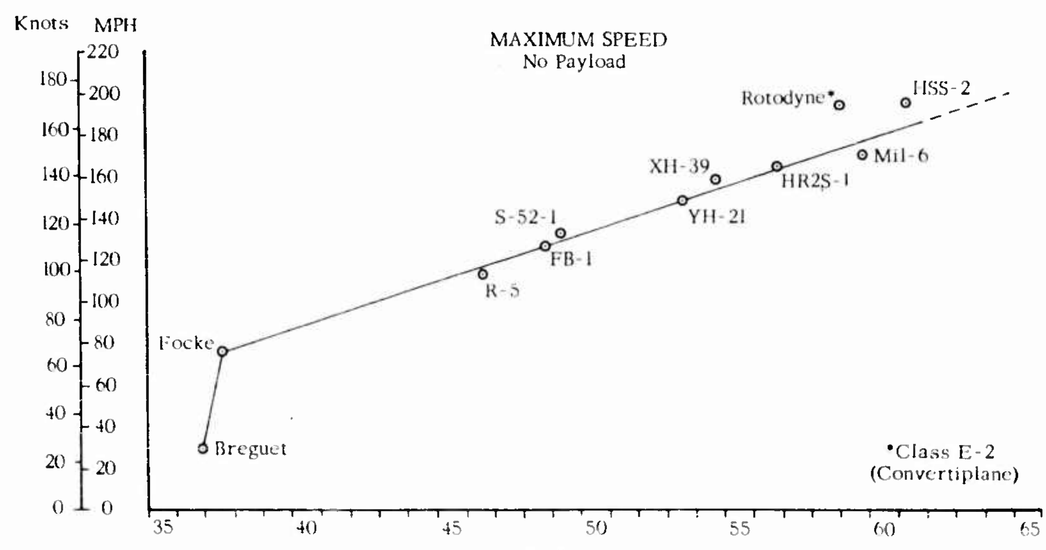

FIGURE 6. HELICOPTER INTERNATIONAL SPEED RECORD

Because of the helicopter's versatility in performing a wide variety of missions, its good safety aspects, low downwash velocities, and low basic noise levels compared to other forms of VTOL aircraft, it is important that the potential performance improvements mentioned be accomplished. The present design study is intended to demonstrate some of these potential performance improvements of the helicopter in terms of an actual preliminary design of an aircraft that could be built and flown in the immediate future. Thus the results presented do not in any sense represent an "ultimate" in development, but rather a practical extension of existing knowledge. However, the helicopter described herein is considered to be a research vehicle because the performance improvement is relatively larger than that normally associated with new designs, and because no specific mission has been defined. This vehicle would be used to investigate the characteristics of the helicopter at high speed, the problem areas that may arise, and to investigate modifications to the basic pure helicopter to provide additional increases in speed, range, or payload.

Sikorsky Aircraft has conducted this study under contract to the U. S. Army Transportation Research Command (Contract DA 44-177-TC-648). Funding of this study has been shared jointly by the Army and Sikorsky Aircraft.

The aircraft is specified to have a conventional single main rotor helicopter configuration and has a minimum design flight speed of 175 knots and a still-air ferry range of at least 1600 nautical miles plus an hour fuel reserve. Flying and handling qualities are to be in accordance with MIL SPEC 8501. Payload is to be sufficient to make the aircraft competitive with existing helicopters.

In addition, the design is required to emphasize the use of existing components to as great an extent as practical, to minimize the eventual effort involved in detail design or development of new components. The design study includes analysis of weights, stresses, stability and control, dynamics and aerodynamics as required to prove that the final design is capable of achieving the desired performance.

SECTION III: DESIGN ANALYSIS

A. SELECTION OF BASIC AIRFRAME

1. Airframes Considered and Rejected

a. General - Only single rotor helicopters were considered according to contract specifications. Previous studies conducted by Sikorsky of the high speed potential of various helicopters have indicated that none of the existing machines powered by reciprocating engines have the potential for achieving the performance required in this study (175 knots, 1600 nautical miles ferry range). Accordingly, the Sikorsky S-51 (H-5), S-55 (H-19), S-56 (H-37), and the reciprocating S-58 (H-34, HSS-1) were eliminated without further consideration.

b. S-59 (XH-39) - This helicopter, shown in figure 7, set a world's speed record of 156 mph in 1954. It was powered by a 400 horsepower Artouste II turbine engine, and incorporated a retractable landing gear system. This aircraft was rejected because of the following considerations:

(1) Only two of these aircraft were built, with a consequent lack of development of components compared to other helicopters as well as inadequate support likely in a future flight research program.

(2) The engine and transmission power capacity are inadequate for the required speed of 175 knots. A different engine installation would be required as well as transmission modifications.

(3) Main rotor solidity is inadequate for the high speed condition, so that a new rotor head design would be required to accommodate additional blades or blades of larger chord.

(4) Because of the small size of this aircraft and the relatively poor specific fuel consumption of likely engine installations, it would be difficult to provide the fuel capacity required for the ferry mission. Payload at short ranges would also be smaller than desired.

c. S-62 - This helicopter, shown in figure 7, is a modernized version of the Sikorsky S-55, with a General Electric T-58 turbine engine and a flying boat hull to provide increased useful load and increased utility. Both 3 and 4-bladed configurations have been developed. This helicopter was rejected for the following reasons:

(1) The parasite drag of the fuselage, with a boat hull and sponsons, is too high to permit either the speed or range required in this study.

(2) The transmission power capacity is limited to approximately 700 horsepower, and while this could be increased to about 800 horsepower by operating at higher rotor rpm than normal, the capacity would still be substantially inadequate for the high speed requirement.

d. HSS-1F - This aircraft, shown in figure 7, is a modification of the HSS-1 (S-58, H-34) incorporating two General Electric T-58 turbine engines instead of a single Wright R-1820 reciprocating engine. Two of the aircraft were built to gain operational experience with the T-58 engine prior to the construction of the HSS-2 (S-61) aircraft, which also incorporates a twin T-58 installation. At the start of the present design study, this configuration was favored for the following reasons:

(1) The basic helicopter is a well-developed machine (over 1200 built) with reliable components and excellent support available.

(2) High payload to gross weight ratio would permit carrying enough fuel to meet the range requirements.

(3) Fuselage is relatively clean, so that only minor modifications to the aerodynamic shape would be required (other than incorporation of a retractable landing gear and rotor head fairing).

(4) The available ranges of collective and cyclic pitch motions and permissible blade flapping angles are greater than for most helicopters.

(5) The twin turbine installation would provide adequate power for the high speed flight condition, as well as permitting good specific fuel consumption on the ferry mission by cruising on one engine at high power.

(6) There existed the possibility of utilizing an existing Navy HSS-1F airframe in the custody of the contractor for constructing the research aircraft.

As a result of the above factors, the HSS-1F airframe was tentatively selected as the best choice for satisfying the contract objectives. The airframe had some disadvantages, of course, which were recognized from the outset, but in the detailed study conducted, some additional defects became apparent.

Disadvantages of the airframe include:

(1) Rotor solidity with the standard 4-bladed rotor is inadequate for the high speed condition. While there is an existing design for a 5-bladed version of this head which could be readily constructed, this would add somewhat to the cost of the vehicle. Furthermore, five S-58 blades, while providing sufficient blade area for the high speed flight condition, providing the weight could be held down to about 11,000 pounds, was still considered marginal for this purpose, and a somewhat higher solidity was desired. A six bladed version of this rotor head is not possible without basic design changes, and the use of existing blade types of wider chord (such as the S-61 blade) would not be satisfactory because of increased centrifugal loads on the bearings. A five bladed S-61 rotor head could be adapted to the S-58 helicopter only with considerable difficulty.

(2) The transmission system has at best only marginal capability for meeting the high speed requirement. The S-58 main transmission has a continuous power rating of approximately 1350 horsepower at an input speed of 2800 rpm (rotor tip speed 727 feet per second for a radius of 28 feet). Modifications considered to increase this capacity were limited to relatively simple changes such as increasing gear widths within the envelope of the present transmission housing to avoid the substantial expense involved in major redesign and subsequent proof testing. With the changes considered feasible, the horsepower capacity could be increased to approximately 1400 at a tip speed of 650 feet per second or 1500 at a tip speed of 700 feet per second. (Performance calculations indicated that the tip speed should be in the vicinity of 650-700 feet per second for the high speed conditions). These ratings are not believed to be adequate for the design flight speed.

While the above power ratings could be exceeded for short periods of time at the expense of a very rapid drop in transmission life, it was considered very undesirable to jeopardize any future flight research program with a marginal component when a better one (S-61 transmission) is readily available.

(3) In addition to an inadequate power capacity in the main gear box, the HSS-1F transmission system is inferior to that of the HSS-2 (S-61) in two other respects: complexity and weight. It is more complex because there is a reduction gear box on each engine and an adjacent but separate coupling gear box to transmit the power of both engines up an inclined shaft to the main transmission. This multiplicity of gear boxes, together with a separate oil cooling system for the lower units, also accounts for the increased weight relative to the S-61 transmission.

The gear box system as used on the HSS-1F also forced some additional complications on the design of the high speed vehicle. The gear box arrangement requires that the engine exhaust be directed downward initially; on the HSS-1F the exhaust of both engines was then brought to the left side of the aircraft, one stack above the other. Use of the region below the engines for exhaust pipes precludes its use for wheel wells in a retractable landing gear system in which the gear folds forward during retraction. The only other retractable gear system considered feasible for this aircraft was one that did not fold up completely within the body envelope, and was, therefore, undesirable from the standpoint of parasite drag.

To avoid the above arrangement, a modification was considered to the gear box system in which the forward section of an S-61 transmission was substituted for the three lower gear boxes used on the HSS-1F. This modification would have saved a substantial amount of weight, and also would have permitted the use of sideward exhausts, freeing the space below the engine compartment for use as wheel wells. This change would have meant additional expense, however, and there was also a question as to whether the exhaust on the pilot's (right) side as well as on the copilot's side would seriously impair the pilot's downward vision.

(4) The structural design of the forward part of the HSS-1F fuselage is not optimum from a weight standpoint because it was simply a modification of an HSS-1 (reciprocating engine) to a turbine test bed. The incorporation of a retractable landing gear on this aircraft would require still more substantial change in the structure, and the achievement of reasonable structural continuity proved to be nearly impossible. The empty weight of the airframe modified to the high speed configuration would, as a result, be excessively high, making both the high speed and long range requirements difficult to achieve. Also,because of the extensive structural changes required by the retractable gear, the large horizontal tail, and the strengthening of the canopy structure to withstand the higher aerodynamic forces, it became apparent that the cost of these modifications would be considerably more than first anticipated.

As a result of the study conducted relative to the above factors, it was finally decided to reject the HSS-1F configuration from further consideration, since it was questionable as to whether the specified performance could be obtained. It is believed that the configuration selected, built around the S-61 powerplant package, will have substantially superior potential for achieving speed, range, and payload, for approximately the same or lower overall cost.

e. S-61 (HSS-2) - This aircraft (figure 7) represents an evolvement from the HSS-1 (S-58), with two General Electric T-58 turbine engines and a flying boat hull for water alighting capability. Its engines and main transmission system have been integrated into a compact and efficient unit above the fuselage. This powerplant package is nearly ideal for meeting the objectives of the present high speed research helicopter design study, and has therefore been selected for this use. The fuselage configuration, however, is unsuitable for the following reasons:

(1) The flying boat hull and sponsons have an excessive parasite drag, and achievement of the specified performance would not be possible.

(2) The clearance provided for downward blade flapping over the tail cone is less than that desired for high speed, since flying qualities studies have indicated that flapping sensitivity to gusts or control inputs increases rapidly with increasing forward speed.

(3) The tail pylon, because of its relatively small size and high sweep angle, does not readily lend itself to the addition of an effective rudder, which is believed to be highly desirable for the high speed research vehicle.

2. General Features of New Airframe Based on S-61 Powerplant System

An artist's drawing of the recommended aircraft is shown in figure 1, and the general arrangement of this configuration is shown in figure 8. A dimensional comparison of the high performance helicopter design with the S-61 (HSS-2) is shown in figure 9. Important features of the high performance helicopter design include the following:

a. Powerplant package is essentially identical to that used on the S-61 helicopter, including twin turbine engines, transmission, and five-bladed main rotor head. Engines are General Electric T-58 turbo-shaft units. Either the -6 version (900 horsepower normal rated, 1050 horsepower military), or the -8 version (1050 horsepower normal rated, 1250 horsepower military) are suitable, with the latter preferred because of the higher power ratings and lower specific fuel consumption.

b. New fuselage design has substantially lower parasite drag than the S-61 fuselage, with smaller height, width, and more stream-lined cross section. Cabin volume is also reduced, but still is adequate for more than 20 troop seats. Cockpit and flight control system are based on existing hardware.

c. Main and tail gear are completely retractable, folding forward and up into the fuselage bottom with no external bumps. Doors cover the gear when retracted, so that no residual parasite drag is incurred.

d. A rotor head fairing system developed by Sikorsky minimizes the parasite drag of the main rotor head and pylon region. This fairing "floats" with the tip path plane and is flexibly mounted on the blades rather than on the head itself, to minimize the size of the blade cutout holes. A blowing boundary layer control jet and a telescoping afterbody, developed in full scale wind tunnel tests (figure 10), complete the system. This fairing system has proved to be essential for achieving the desired performance.

e. A rotor diameter of 56 feet was selected, equal to that of the S-58 helicopter (HSS-1, H-34), instead of the 62 foot rotor used on the S-61. This diameter is simply achieved by cutting 3 feet from the tip of the S-61 main spar. The smaller diameter has adequate performance capability, and has the advantages of increasing the transmission power capacity (since the transmission is torque-limited, and a higher rpm is used for the smaller diameter) as well as permitting incorporation of an aerodynamically more suitable tail design for the high speed condition without increasing body length or upsetting center of gravity position.

f. An effective fixed vertical stabilizer and rudder has been provided by reducing the sweep of the tail pylon. The rudder may be used to offset main rotor torque at high forward speeds to compensate possible loss of tail rotor capability. This tail is also desirable for modifications of the research vehicle into configurations other than the pure helicopter.

g. A large horizontal tail provides inherent longitudinal stability at all speeds above 50 knots. An elevator is incorporated to permit longitudinal trim control, a longitudinal control stick force gradient similar to that of a fixed wing airplane if desired, and longitudinal control for configuration modifications beyond that of a pure helicopter.

h. Blade flapping clearance over the tail cone has been increased relative to the S-61, to allow for the increased flapping sensitivity expected at high speeds. This was accomplished by lowering the tail cone and by the reduction of the blade radius by three feet.

i. A modified engine inlet design has been incorporated to reduce its drag at high forward speeds. An engine exhaust deflector has also been incorporated to recover most of the engine residual thrust.

j. Fuel for normal flight operations is carried below the cabin floor. Fuel for the ferry mission is carried in a special rubber fuel cell inside of the cabin. No external tanks are required.

k. This design utilizes many existing components not requiring redesign, development, or extensive modifications, and thus minimizes the cost of detail design and construction of the research vehicle. These components include:

(1) S-61 powerplant system, engine and main transmission
(2) S-61 main rotor head
(3) S-61 main blade retention system, spar, trailing edge pockets, balance features (actual blade identical to S-61 blade except for length and twist)
(4) S-61 tail rotor
(5) S-61 tail gear box
(6) S-61 input and output end castings and bearings for intermediate gear box
(7) S-61 hydraulic system
(8) S-61 electric system (simplified to requirements of research helicopter)
(9) S-61 oil cooler
(10) S-61 tail rotor drive shaft components
(11) S-62 cockpit configuration, including seats, controls, instrument panel, etc. Structure of S-62 cockpit requires reinforcement to carry aerodynamic loads on window panels.
(12) Flight controls are a combination of existing S-62 and S-61 hardware.

B. ESTIMATION OF PARASITE DRAG

1. Importance of Parasite Drag

Previous Sikorsky studies, such as that reported in Reference 2, have indicated the fundamental importance of achieving low parasite drag for the attainment of both increased speed and range capability of the helicopter. The aerodynamic efficiency of the rotor itself is reasonably good, having typical equivalent lift-drag ratios of 10 to 12. The overall effective lift-drag ratio for present helicopters, however, is only about 4 or 5, the difference being predominantly the high parasite drag of the fuselage (including landing gear, rotor head, etc.). Attainment of a 175 knot speed with a pure helicopter would be extremely difficult with present parasite drag characteristics, and achievement of a still-air ferry range of 1600 nautical miles plus a one hour fuel reserve would be virtually impossible. Accordingly, the reduction of parasite drag is considered the most important single factor in the design of a vehicle to achieve the required performance.

The drag of most helicopters can be substantially reduced by adopting standard fixed wing aircraft practice . Items such as the landing gear, engine inlets, and exhaust stacks are examples of drag producing components that are subject to standard cleanup programs. One large drag item of the helicopter, however, is not subject to standard treatment: the rotor head area. Conventional streamline fairing shapes are not possible because of the rotating components including the blades which are also free to feather, flap, and lag. The rotor head requires an aerodynamic treatment, however, because of both its own appreciable drag and an interference drag usually found between the head and the pylon body. Results of a newly developed rotor head fairing are discussed below.

2. Rotor Head Fairing Test Results

a. Description of Model As part of a Sikorsky sponsored research program to reduce parasite drag, a wind tunnel test was conducted to determine the merits and operating characteristics of a rotor head fairing designed to "float" with the tip path plane. The concept of this fairing is to mount it on the blades with flexible supports, rather than on the rotor head itself, so that the size of the blade cutout holes will be minimized. Previous efforts to mount fairings rigidly to the rotor head have resulted in blade cutout holes so large that no aerodynamic benefit was obtained. With the "floating" fairing, the largest blade motions (coning, first harmonic flapping, and mean lag angle) are automatically accommodated, so that only blade feathering and higher harmonics of flap and lag require clearance provisions.

The rotor fairing, based on previous tests on a model scale, was built full size out of fiberglas and tested under actual operating conditions. Photographs of this model are shown in figure 10. The body of this model represents approximately the entire helicopter main pylon including engine, transmission, and auxiliary equipment sections. The model incorporated an actual S-61 transmission and rotor head assembly, and was driven at normal S-61 operating rpm by an electric motor attached to the tail rotor drive shaft. Stub blades were mounted on the rotor head, and were feathered in a manner similar to that encountered in high speed flight (the swash plate was locked to provide a collective pitch of 16 degrees measured on the blade cuffs and a longitudinal cyclic pitch of 12 degrees). The model was tested at various angles of attack, and various combinations of rotor coning and tilt were obtained by a "May-pole" arrangement wherein cables from each blade tip were attached to a common rotating connection at selected locations on the tunnel ceiling.

Another feature of this model is the incorporation of a blowing boundary layer control jet on the cylinder between the fairing and pylon body (figure 10). Previous small scale tests indicated the need for flow control in this region to avoid a large interference drag. The model also incorporated a sliding rigid seal to close the bottom of the fairing, and a telescoping afterbody, with felt as the bearing material between the afterbody and the rotor head fairing, to provide a smooth airflow path to the rear of the model.

b. Test Results Tests were conducted in the United Aircraft 18-foot wind tunnel to determine both operating characteristics and performance up to a speed of 157 knots, the approximate maximum tunnel velocity with this size model. The test was considered successful in that operating characteristics were satisfactory and performance results justified the parasite area assumption made in the performance analysis (Section III C).

Functionally, the model performed very well. The suspension, consisting of a series of rubber shock chords mounted radially from the blade feathering axis, provided adequate stiffness for support of the aerodynamic loads on the fairing up to the maximum test velocity but was flexible enough to permit blade feathering and higher harmonics of blade motion without excessive restraint. Despite a small out-of-round condition of the lower fairing half, producing a one-per-revolution vertical oscillation of approximately one-half inch, the afterbody, which was spring loaded to stay in contact with the fairing, followed this motion without difficulty. The felt seal between afterbody and fairing was very satisfactory, showing practically no signs of wear at the end of the test program. The rubber seals used on the blade cutout holes were not completely satisfactory because of a lack of tear resistance, but this did not affect the mechanical operation of the model.

Several boundary layer control (blc) nozzle configurations were tried; the best configuration tested was a slot one-half inch wide (the widest jet width available) by 12 inches high (extending from the top of the pylon body to the bottom of the fairing for a 6 degree coning angle) on each side of the model. The jet velocity was more or less uniform from top to bottom; time did not permit experimentation with other distributions that might have been more efficient. It is also not known if the left side of the model requires less jet velocity than the right side; this possibility exists because a "Magnus Effect" due to rotation of the fairing may help the flow control to some extent on the left side. Additional tests might therefore show some benefits to be obtained by variations in the boundary layer control system. In this respect the results of the present test are believed to be conservative. The results are also believed to be conservative because a more streamline fairing shape is proposed for the high performance helicopter design than was tested in the wind tunnel. A thinner shape, particularly near the rim of the fairing, may be used on the proposed design because the blade folding hardware outboard of the feathering bearing, which was present on the wind tunnel model, will not be used.

Figure 11 shows the experimental drag results for this fairing-pylon model for a body angle attack of zero degrees (shaft angle -4 degrees), which is the approximate trim angle of attack at the high speed flight condition indicated by the flying qualities study (Section III E). Results are given in terms of parasite area as a function of nozzle pressure ratio. The lowest curve, labeled "effective external drag," represents drag measured by the balance (corrected for model blockage, support tares, and blade stub drag), modified by the addition of the ram drag corresponding to the blc system mass flow. This ram drag was not present in the tunnel tests, but would exist in flight. This bottom curve represents the significant drag of the model as far as the main rotor is concerned since this drag force (plus the drag of other body components) is the force which the main rotor must overcome. It may be seen that very substantial reductions in rotor propulsive force required can be obtained, depending on nozzle pressure ratio used. The middle curve of figure 11 is the actual model external drag, obtained by adding the net thrust of the jet (gross thrust minus ram drag) to the effective external drag. The top curve, labeled "total equivalent drag," represents the significant result of the test as far as overall aerodynamic performance is concerned; it is obtained by adding the parasite area equivalent of the pumping power required for the boundary layer control system (assuming an airflow system efficiency of 75 percent)to the effective external drag.

The total equivalent drag curve exhibits a minimum point at a pressure ratio of approximately 8, for which the indicated total equivalent drag corresponds to 6.5 square feet of parasite area, and the effective external drag is 5.0 square feet. Assuming operation at 175 knots at sea level, the actual jet total pressure is 5.8 psi above atmospheric at this point, the mass flow is 5.2 pounds per second, and the pumping horsepower required is 94. The size of the blc system components may be reduced if desired by the use of a nozzle pressure ratio to the left of the minimum point; operation at a pressure ratio of 6 increases the total equivalent drag from 6.5 to 6.7 square feet, but reduces the pumping horsepower to 61.

It is of interest to note that without the boundary layer control (zero pressure ratio) the drag of the system is higher than for the standard unfaired head, due to severe aerodynamic interference between fairing and pylon.

c. Estimated Improvements It is believed that the test results can be improved by a significant amount in designing the fairing for the high speed helicopter. Tests were previously conducted at a smaller scale on a series of fairing shapes, and it was determined that a sharper-edged shape had substantially less drag than the relatively blunt shape used in the present investigation. The blunt shape was required for the selected fairing diameter to provide adequate clearance for the rotor head components used in the test. These components included blade folding provisions outboard of the feathering bearing which are not part of the present proposal. By utilizing the "commercial" S-61 rotor head without folding, a substantial amount of rotor head weight is saved and the effective size of the rotor head is decreased permitting a thinner fairing shape and a lower drag. Based on the previous model tests, it is estimated that 0.75 square feet of parasite area can be saved. Additional reductions in drag should be possible by better sealing at the blade cutout holes and the bottom sliding seal mechanism. The rubber seal used at the blade cutout holes during the tunnel test was not satisfactory because of a lack of tear resistance of the material employed. When the seals were partially ripped, as they were for the majority of the test points, the centrifugal action made them protrude into the external airflow, as well as allowing air leakage in and out of the fairing. Lack of an adequate seal had previously been determined to cause an appreciable drag increment. In addition, the bottom sliding seal mechanism allowed an appreciable air leakage in or out of the fairing along the surface of the cylinder. This air leakage, as revealed by tuft patterns, was quite noticeable just ahead of the blc jet and acted as a flow spoiler in this region. Both of these air leakage conditions could be eliminated, by use of a tear-resistant material at the blade cutout holes, and by an improved bottom fairing seal.

It is conservatively estimated that at least 1.0 square foot of parasite area could be saved by the use of a thinner shape and by proper sealing of the fairing as discussed. As mentioned previously, it is also believed that improvements are possible in the boundary layer control system. Any such improvement would have to come from a second experimental investigation and no estimate is attempted here. It should be noted that the above estimates of possible improvements are not required to justify the drag assumptions made in the performance section of this report.

3. Estimate of Total Aircraft Parasite Drag

Results of the rotor head fairing test described above are used for the fairing and pylon without accounting for the possible additional drag reduction as discussed. Other items are estimated by conventional techniques. It is assumed that the attention devoted to detailed aerodynamics during the construction of the research helicopter will be comparable to that employed in modern, moderate-speed fixed wing aircraft.

Item	Parasite Area, Square Feet
Rotor head fairing and pylon (upper part of fuselage)	6.50 (includes blc power equivalent. Effective external parasite area 5.0 square feet.)
Basic main body	2.20
Interference due to decrease of wetted area for combination of above 2 bodies	-0.30
Surface irregularities, (rivets, lap joints, gaps)	0.35
Windows, canopy irregularities, sliding cockpit hatches	0.60
Engine inlet installation	0.30
Engine exhaust nozzles	0.15
Tail surfaces including tail rotor transmission fairing	1.20
Induced drag of vertical tail at 175 kt. assuming zero tail rotor thrust	0.70
Tail rotor blade profile drag & faired head	0.70
Miscellaneous inlets, outlets, etc.	0.50
	12.90 (includes blc power equivalent. Effective external parasite area 11.4 square feet.)

The parasite area used in the performance calculations (Section III C) was 13.0 square feet. This assumption is justified by the above figure of 12.90 square feet, which includes the drag equivalent of the blc system power required. The actual effective external drag (rotor propulsive force required) is 11.40 square feet. These figures do not reflect possible improvements in the rotor head fairing system; these improvements were estimated to amount to at least 1.0 square foot, not counting possible improvements in the boundary layer control system. The other parasite drag items were also conservatively estimated; it is believed that the total figures given above can be achieved without difficulty, and can perhaps be significantly reduced with proper attention to detail design. The value of 13 square feet is slightly less than one-half of the parasite area of the S-61 helicopter, and only slightly more than one-third of the parasite area of the S-58.

No variation of parasite area with angle of attack is given because of the small range of trimmed angle of attack for level flight indicated by the flying qualities study. Wind tunnel tests of various fuselage models has shown that this assumption is reasonable. At angles of attack corresponding to high rates of climb or to autorotative descent, the parasite areas would be somewhat higher than those presented.

C. PERFORMANCE CALCULATIONS

1. Method of Calculation

The procedure used for calculating main rotor performance in forward flight is shown schematically in figure 12. This method, which utilizes an IBM 704 electronic digital computer, has no restrictions on advance ratio or forward speed and no small angle assumptions. It incorporates experimental two-dimensional airfoil data, including stall and Mach number effects, obtained with Sikorsky production blade sections having an NACA 0012 airfoil. Zero-lift airfoil drag coefficients were increased to at least .010 for use in the calculation, compared to test values of .006 to .008 for low and moderate Mach numbers, to account for possible leading edge abrasion on a blade in service. Performance calculations are believed to be conservative for this reason.

The calculation proceeds from an arbitrary starting point for any given combination of forward speed, tip speed, inflow ratio, and collective pitch. Initially a rotor blade is assumed to be in some particular flapping condition at zero azimuth position. This starting assumption defines the instantaneous section angle of attack distribution along the blade and thereby permits evaluation of all of the instantaneous blade forces. The subsequent time history of the blade may then be calculated for small increments in azimuth position (10 degrees is usually used), and the blade is allowed to go through a number of complete revolutions in this manner until flapping convergence is achieved. Flapping convergence occurs when the blade motion repeats exactly (within a small mathematical tolerance) for two successive revolutions. The converged motion and the integrated rotor forces and moments are independent of the initial assumption of blade motion.

This procedure has provided excellent correlation with available helicopter flight test data, including the speed record of 192.9 mph established by the HSS-2. The procedure has also been evaluated under another Army contract study, reference 4, conducted with wind tunnel tests of dynamic model blades over a wide range of tip speeds and advance ratios. Good correlation has been obtained at rotor lift values below calculated retreating blade stall, and at higher lifts the theory is conservative (experimental rotor performance superior to theoretical). The calculation procedure thus has considerable justification for use in the design study. Typical theoretical results obtained with this performance method at high speeds and advance ratios are presented in reference 2.

Retreating blade stall is assumed to correspond to a limiting value of integrated torque coefficient due to airfoil drag for one blade at any azimuth position on the retreating side. The limiting value corresponding to a moderate degree of stall is $bC_{Qd}/\sigma = .004$, equivalent to a mean relative velocity of $0.4\,\Omega R$ and a section drag coefficient of 0.10 (corresponding to a blade section angle of attack slightly above stall at most local Mach numbers). This parameter is a sensitive indication of the amount of stall on the retreating side of the disk. As mentioned above, it should be a conservative limit.

Hovering performance is calculated by a numerical blade element-momentum method utilizing the same airfoil data as the forward flight calculation, and the IBM 704 digital computer. Tail rotor power required in hovering is independently calculated by the same technique used for the main rotor. Total power is the sum of main and tail rotor power plus 100 horsepower for auxiliary power requirements and 2 percent for transmission losses.

In forward flight the main rotor power required is divided by an efficiency factor of 0.90 to account for tail rotor power, transmission losses, etc. This factor has been found to provide a reliable estimate of total power required at speeds of 100 knots or more. Allowance for boundary layer control air pumping power is not included in this efficiency factor because the parasite area of 13.0 square feet assumed for all forward flight calculations includes the parasite area equivalent of the pumping power (see Section III B). The actual power distribution would be less to the main rotor and more to the auxiliary mechanisms than calculated; however, the total power is the same.

It is assumed that the engine airflow does not contribute a horizontal force that must be accounted for in the performance calculation. The validity of this assumption may be checked by examination of figure 13, which shows the variation with forward speed of gross jet thrust and ram drag for one T-58-8 engine at normal rated power. The lower half of figure 13 presents the forces in pounds; the upper half shows the same data in terms of the equivalent thrust or parasite areas. Three curves of gross jet thrust are given for exhaust angles of 0, 30, and 60 degrees from the line of flight. The T-58 engine is built with an exhaust angle of 60 degrees from the engine shaft axis, and as installed in the S-61 helicopter the exhaust angle is also 60 degrees from the line of flight. One-half of the engine's gross jet thrust is lost as a result. This is not particularly important at low or moderate speeds, but at the design speed of the present high performance helicopter study (175 knots), the ram drag would exceed the gross jet thrust by a very significant amount. Accordingly, it was decided that short exhaust stack extensions, with vanes to deflect the exhaust efficiently to a 30 degree angle, would be required in order to regain most of the lost jet momentum. While this incurs some additional external drag due to the increased frontal area, the gain in jet thrust is considerably more important.

With the use of an exhaust angle of 30 degrees, the assumption that the engine airflow forces may be neglected in the performance calculations is a conservative one, particularly at speeds lower than 175 knots.

2. Selection of Rotor

The main rotor design represents a compromise between performance requirements and blade stress and vibration considerations. This choice is discussed more fully in Section III D. The selected rotor has an adequate performance capability for all speeds from hover to 175 knots, and has the following geometric characteristics: rotor radius 28 feet, five blades, blade chord 18.25 inches, solidity 0.086, -4 degrees of linear twist, and NACA 0012 airfoil section. Tip loss factor was taken as 0.97 for the performance calculations. The full blade chord was assumed to extend inboard to the rim of the rotor head fairing (44 inch radius = 13 percent rotor radius).

Reference 5 has indicated that there might be a performance advantage to be gained by leaving a "root cutout" region where the chord of the blade is reduced by eliminating trailing edge pockets near the root. To check on this possibility, calculations of rotor performance were made with a root cutout extending to 25 percent rotor radius. Experimental two-dimensional lift and drag data obtained with an actual Sikorsky blade spar section, with an angle of attack range from 0 to 360 degrees, was utilized in the calculation. The spar, which represents approximately the forward 45 percent of the full airfoil chord, was modified by the addition of a semicircular trailing edge cap to reduce drag near 180 degrees angle of attack. The study indicated that if the forces on the root cutout region could be entirely eliminated at the design level flight condition of 175 knots, the power required would be decreased about 75 horsepower. With the actual spar forces included, however, the power required would be reduced by a negligible amount (about 20 horsepower, within the accuracy of the calculation). For this application there is apparently no appreciable benefit to be derived from root cutout. The reasons for the apparent disagreement with reference (5) are believed to be the relatively large chord and consequent drag of the spar section employed and the relatively low tip path plane tilt required. Because of the low fuselage parasite drag, the inclination of the rotor resultant force vector is only 5.0 degrees forward at 175 knots for the normal gross weight of 13,600 pounds. Tip path plane inclination at this condition is 6.6 degrees forward. The benefits of root cutout would be greater for more forward rotor tilts. The blade planform geometry shown in figure 8 is a compromise between the above two cases, with a non-structural tapered trailing edge section between the rotor head fairing and the 22 percent radius station, providing a partial cutout close to the center of the rotor where it is of greatest value.

The tail rotor is the same as that used on the S-61 helicopter, having a diameter of 10.0 feet, five 7.35 inch chord blades, solidity 0.195, and zero twist. For a given main rotor horsepower the torque is reduced on the high speed helicopter design because the rotor rpm is increased compared to the S-61. Effective tail length is reduced by approximately the same factor so that tail rotor thrust required is about the same. The S-61 tail rotor should be entirely satisfactory from hovering up to at least 135 knots (the approximate design speed of the S-61). At higher speeds, tail blade edgewise stresses might become excessive if full thrust required to counteract main rotor torque is developed, because of the absence of lag hinges. Consequently, an effective vertical fin and rudder have been provided so that the fixed surfaces are capable of counteracting full main rotor torque at 175 knots. Wind tunnel experience with S-61 tail blades has indicated that no trouble should be anticipated at the high speed condition if tail rotor thrust is kept to a moderately low value.

3. Calculated Performance Results

a. Forward Flight Performance The calculated forward flight performance results are presented in figures 14-17. The rotor tip speed used for the majority of these calculations was 650 feet per second, which is high enough to provide adequate lifting capacity for both the ferry mission and the high speed flight condition at relatively low gross weights, and low enough to avoid adverse Mach number effects on the advancing blade. At 175 knots substantially more payload can be carried if the tip speed is increased to 700 feet per second (advancing tip Mach number 0.89); therefore, additional data are presented at higher tip speeds.

Figures 14, 15, and 16 present engine power required versus forward speed and gross weight for altitudes of 0, 5000, and 10,000 feet, respectively. The blade stall limit, shown by the dashed line, indicates that the maximum gross weight at sea level (figure 14) varies from over 20,000 pounds at 75 knots down to 14,800 pounds at 150 knots and 11,300 pounds at 175 knots for a tip speed of 650 feet per second. At this tip speed, the rotor is not capable of absorbing even the normal rated power of two T-58-8 engines. At a tip speed of 700 feet per second the permissible gross weight increases along with an increase in power required. In figure 14 a shaded square symbol is shown at 175 knots on the curve of normal rated power available. This point represents a gross weight of 13,600 pounds, the maximum value permitted by the normal power limit at a tip speed of 700 feet per second. This point has been selected as the basic design point of the high performance helicopter. At the military power rating of the engines the forward speed may be increased to 182 knots at the same gross weight and tip speed (upper shaded square). This point is also the approximate theoretical retreating blade stall limit at this tip speed and gross weight. A value of 182 knots is thus the maximum speed available at full nominal gross weight and thus with full nominal payload (two tons).

Maximum speed without payload is also indicated on figure 14 for interest. This maximum speed is 195 knots, identified by the shaded triangle. Gross weight is 8700 pounds, equal to the operating weight empty (Section IV B) plus a small allowance for fuel. Tip speed was optimized at 675 feet per second for this condition; a higher tip speed requires excessive power because of Mach number effects while a lower tip speed would result in excessive retreating blade stall. It should be noted that for this flight condition the profile drag penalty for possible leading edge abrasion was not applied, as it was considered reasonable to expect smooth blades for a high speed demonstration of this kind. Minimum blade section drag coefficients used, however, were not less than .008 even at moderate Mach numbers. This value is still relatively conservative for the NACA 0012 airfoil at the operating Reynolds numbers.

The effects of tip speed on rotor lift capability and power required are shown somewhat more explicitly in figure 17 for forward speeds of 150 and 175 knots at sea level. Increasing tip speed from 650 to 700 feet per second increases maximum lift from 14,800 pounds to 17,700 pounds at 150 knots and from 11,300 pounds to 14,400 pounds at 175 knots. Similar increases would be expected at other speeds, and altitude capability would also be improved. The upper half of figure 17 shows the power requirements corresponding to the lower half of the figure. While some of the power increase with tip speed is due to induced drag associated with the increased lift, the increase is predominantly the profile power due to increased tip speed and advancing tip Mach number.

Also shown in figure 17 are the selected design points at 175 knots and an additional design point at the reduced speed of 150 knots. This latter point is of interest because it provides the approximate maximum productivity (discussed in Section IV E) of the basic high performance helicopter. At 175 knots, the gross weight for a tip speed of 700 feet per second is limited by available power to 13,600 pounds (shaded square). This gross weight is roughly 5 percent less than the rotor capability. At 150 knots there is no power limitation but it was decided that the gross weight should be limited to 17,000 pounds (shaded diamond), also roughly 5 percent less than the theoretical stall limit, as an additional conservative factor.

b. Hovering Performance The hovering performance of the rotor for a tip speed of 650 feet per second at various altitudes is shown in figure 18 in terms of main rotor lift versus total engine horsepower required, including calculated tail rotor power, 100 horsepower for auxiliary items and 2 percent for transmission losses. Hovering ceilings derived from these data are shown in figure 19, assuming that gross weight is 3 percent less than rotor lift because of vertical drag of the fuselage. With two T-58-8 engines operating at military power, a gross weight of 20,000 pounds may be hovered out of ground effect at sea level. At normal rated power gross weights up to 18,000 pounds may be hovered out of ground effect at sea level. This limit of 18,000 pounds is considered the maximum gross weight that would provide satisfactory operation under normal conditions. At a gross weight of 13,600 pounds, the design value for 175 knots cruise speed, hovering out of ground effect is possible up to 12,000 feet altitude with normal rated power. This performance is outstanding compared to current helicopters because of the high power to weight ratio of the high performance helicopter at moderate weights.

With one engine inoperative, a maximum gross weight of 11,500 pounds may be hovered out of ground effect at sea level. According to Reference 6 rotor lift is increased approximately 30 percent by ground effect for a mean rotor height 0.5 rotor radius above ground, corresponding to the wheels-on-ground condition for the subject helicopter design. A more conservative increase of 20 percent, obtained with the wheels about 5 feet off the ground is applied to the hovering-in-ground-effect case shown in figure 19. Maximum gross weight permitting hovering in ground effect at sea level is indicated as approximately 14,000 pounds. Therefore, this aircraft will have excellent one-engine-out characteristics at moderate gross weights.

c. <u>Ferry Range Analysis</u> The maximum (ferry) range with this aircraft is obtained by cruising at moderate speeds on one engine at a relatively high power setting to take advantage of the better engine specific fuel consumption available at high power levels. Either a running takeoff on one engine may be made, or a hovering takeoff is possible with two engines, one of which is shut down when cruising speed is attained.

Optimum flight conditions for a given gross weight are defined as the combination of forward speed and altitude giving the maximum specific range (nautical miles per pound of fuel, equal to forward speed in knots divided by the product of engine horsepower and specific fuel consumption). These optimum flight conditions are shown in figure 20 as a function of aircraft gross weight. Optimum cruise speed is approximately constant at 125 knots up to a gross weight of about 16,000 pounds, above which the cruise speed is limited by blade stall and/or the power available (normal rated) of one engine. Optimum cruise altitude would be above 10,000 feet at low gross weights, but was arbitrarily limited to 10,000 feet to avoid oxygen or pressurization considerations. At higher gross weights the optimum cruise altitude is reduced because of increased induced power, and at about 17,000 pounds or above, sea level cruise is desired.

The specific range corresponding to the flight conditions shown in figure 20 is presented in the lower half of figure 21, and calculated total ranges for various combinations of gross (initial) weight and final weight are shown in the top half of figure 21. For a final weight of 9100 pounds, which is the estimated value including an hour reserve fuel at the end of a ferry mission, the required still-air range of 1600 nautical miles may be attained with a takeoff gross weight of approximately 15,600 pounds (circled point), well within the capability of this machine. For an initial gross weight of 18,000 pounds, which will allow hovering at sea level on normal rated power (two engines), the maximum ferry range is indicated to be 2080 nautical miles plus the one hour reserve (squared point). This figure is taken as the nominal ferry range of the aircraft. The range could be extended by allowing initial gross weights above 18,000 pounds, although the hovering performance would be compromised. For a takeoff weight of 20,000 pounds, a ferry range in excess of 2300 nautical miles would be obtained.

d. Normal Mission Range Analysis The range capability of the high speed helicopter at 175 knots at sea level is shown in figure 22 for various combinations of initial and final weights. The specific range figures shown are calculated on the assumption that rotor tip speed is varied from 700 feet per second at a gross weight of 13,600 pounds (normal power limit) to 650 feet per second at 11,300 pounds (stall limit) and constant at 650 feet per second at lower weights. This tip speed range is well within the normal rpm range provided by the free power turbine of the engines.

The range capability at 150 knots cruise speed, also at sea level, is shown in a similar manner in figure 23. Tip speed is 650 feet per second up to 14,800 pounds and increased to 700 feet per second at higher weights.

An analysis of payload-range characteristics and productivity for various speeds and stage lengths is presented in Section IV.

D. BLADE STRESS AND VIBRATION STUDY

1. Method of Calculation

Prediction of blade vibratory stresses is a key element in the probable success of any high speed rotary wing vehicle. Without an adequate analysis of these vibratory stresses, the useful fatigue life of the blades could conceivably be so low as to render the aircraft completely unusable at high speeds. Experience has shown that critical blade stresses tend to increase rapidly with speed. It was expected that production blades which are completely satisfactory at present helicopter speeds would not be satisfactory for the high speed vehicle; subsequent analysis has shown this to be the case to meet the high performance requirements of the present design study. The normal advancing tip Mach number limits and the retreating blade stall limits must be extended simultaneously. This puts an unusual burden on both the performance and blade vibratory stress predictions, since both are dependent on calculated blade load distributions. Furthermore, there is evidence that the elastic deflections of the blade during rotation change the angle of attack distributions and the resultant aerodynamic loadings by an appreciable degree. Consequently, the standard available methods for calculating blade stresses are not believed to be adequate for the high speed helicopter.

The analysis method used in the present study represents an extension of the advanced performance analysis discussed in Section III C. Like the performance calculation for the rigid blade, the time history of blade motion is computed from an arbitrary starting point over a number of successive revolutions until the blade motion "converges" on its equilibrium pattern. The blade is represented mathematically by a series of rigid segments connected by pin joints and springs to provide bending stiffness equivalent to the actual flexible blade. This system is shown schematically in figure 24. Each of these segments must converge with respect to both angular position and angular velocity within a small tolerance before the calculation is said to be converged.

This calculation incorporates experimental airfoil section data, including stall and Mach number effects. Because the angle of attack distributions include the components due to blade bending, the changes in load distribution due to blade flexibility are automatically included. Aerodynamic damping of the bending vibrations is also inherent in the calculation; no additional damping factors are required. The calculation is presently limited to bending in the plane of flapping; the blade is assumed to be rigid torsionally and in the in-plane direction. The blade spars considered are very much stiffer edgewise than in the flatwise direction, and experience has shown that the edgewise vibratory stresses are generally considerably lower than the flatwise stresses at high speed, so that the above assumption of bending in the flapping plane only should be a reasonable one. Most of the calculations in the present study were conducted with a six-segment representation of the flexible blade, with each segment divided into two parts for the computation of aerodynamic forces. While this seems like a small number, six segments can quite accurately represent a second-mode bending shape (such as shown in figure 24). The natural frequency of this mode is on the order of five cycles per revolution for the rotor selected, whereas the significant stresses are the quasi-static response of the blade to the applied airloads, which are primarily first and second harmonic in nature. Check calculations with an 8-segment program for the high performance helicopter at its high speed design condition showed only minor differences in stresses and root end shears (up to the sixth harmonic) compared to those calculated with the 6-segment program. Because of a considerable difference in the cost of the two programs, the 6-segment version was used. This calculation procedure utilized a Philco S-2000 "Transac" electronic digital computer.

2. Correlation of Theory and Experiment

To demonstrate the validity of the calculation procedure, results of correlation of the theory with experimental results are shown in figures 25-28. Figures 25 and 26 show time histories of blade flatwise bending moments obtained from wind tunnel tests of a one-eighth scale dynamic model of the S-56 main rotor for two spanwise blade stations and three forward speeds, and calculated bending moments for the same conditions. It should be noted that the bending moments, measured with strain gages bonded on the blade spar, are given in terms of full-scale values. These tests are part of another Army contract study, Reference 4. Test conditions selected for this comparison were for approximately the same lift for the three speeds, but with rotor propulsive force increasing with speed, to represent as nearly as possible the conditions corresponding to the high performance helicopter study. In general, both the wave form and amplitude of the bending moments were very well represented by the calculation. The amplitudes of the vibratory bending moments (one half of the peak-to-peak value) derived from figures 25 and 26 are shown in figure 27 as a function of forward speed. This chart shows that the relative magnitudes of the bending moments at the two spanwise stations as well as the increase in vibratory bending with forward speed are properly predicted by theory. The theory predicts somewhat higher magnitudes than the experimental results, possibly as a result of an inadequate set of low Reynolds number airfoil section data curves used in the calculation.

Another correlation of theory and experiment is presented in figure 28, which shows the deflection of the blade tip relative to the root of a blade having -14 degrees twist on an S-56 helicopter in flight at 105 knots. The deflection was measured photographically by means of a high speed movie camera fixed to the root end of the blade outboard of the flapping hinge; the deflection shown is due to blade bending only. Again, the calculated result is very similar with regard to both the wave form and amplitude.

3. Allowable Blade Stresses

The allowable vibratory blade stresses are a function of the specific spar design and the fatigue life demanded. Preliminary fatigue life data obtained on a number of S-61 blade spar specimens are shown in figure 29. The "mean curve" shown represents the average time to failure of the specimens tested; the "working curve" shows the statistically safe values used to determine retirement life of the blade. If a blade life of 500 hours is selected as a minimum desired value at the high speed flight condition to permit a thorough operational evaluation, then roughly 7 million cycles will be accumulated, and the maximum permissible vibratory stress (combined stress at a rear corner of the spar), is about 5900 psi. Experience has shown that the flatwise bending stress is generally on the order of at least two-thirds of this combined stress, so that the allowable flatwise vibratory stress was initially assumed to be limited to about 4000 psi. Check calculations of edgewise stresses at the high speed condition have shown that this should be a conservative limit, and blade life should be substantially more than 500 hours (see below).

4. Selection of Rotor

Numerous calculations were made with the blade bending analysis technique described above in order to select a rotor that is satisfactory from the standpoint of blade stresses as well as performance. Blades considered included S-56, S-58, and S-61 types, the last being selected as most compatible with the S-61 powerplant system as well as satisfactory from the other standpoints. The various calculations have established several fundamental variables that affect blade stresses. These are illustrated by the typical results shown in figures 30-34. Figure 30 shows that at 175 knots blade twist is a major variable in controlling vibratory stress amplitudes (defined as one-half the difference between maximum and minimum stresses around the azimuth for a given radial position). The stresses at all radial positions are reduced substantially by reducing blade twist from the normal value of -8 degrees (the negative value indicates that tip pitch is less than root pitch) toward zero. The same data are shown in figure 31 as a function of blade radius and show that the distribution along the blade changes with twist, the maximum occurring outboard at about 2/3 radius for high twist, and inboard at about 20 percent radius for low twist. The reason for a relatively high inboard stress at all twists is believed to be the relatively high percentage of total blade weight concentrated near the root.

Other effects of blade twist are shown in figure 32. With respect to rotor performance, the upper curves of figure 32 indicate that twist is desirable because it reduces horsepower required and helps to avoid retreating blade stall. Twist, therefore, is beneficial to rotor aerodynamic efficiency at this flight condition (level flight at 175 knots). This is in contrast to the effect of twist on stresses, where twist is undesirable. It should be noted that the performance results in the upper half of figure 32 were obtained with airfoil data corresponding to relatively "clean" blades; the performance differs slightly from results presented in Section III C for this reason.

The effect of twist on the vibratory vertical shears at the rotor head are shown in the bottom half of figure 32, for level flight at 175 knots. The first harmonic vibratory shear is not shown as this is a function of tip path plane tilt relative to the shaft and therefore a function of how the aircraft is trimmed. The higher harmonics are essentially independent of trim. It may be seen that reducing twist is beneficial with respect to reducing vibration. It is also significant that the magnitudes of the vibratory shears drop rapidly with increasing harmonic order. The 5-bladed high performance helicopter, for which the 4th, 5th, and 6th harmonics are important, may be expected to have a much lower vibration level at high speed than a 2- or 3- bladed design.

In addition to the various linear twists considered, several non-linear twists were investigated in an attempt to combine the performance advantages of high twist with the stress benefits of low twist. The particular configurations studied, however, did not show that such a combination of advantages could be achieved.

Another variable affecting blade vibratory stresses is rotor tip speed. Figure 33 shows the calculated stresses over the tip speed range of interest, 650-700 feet per second, for three different twists. Increasing tip speed is beneficial with respect to vibratory stresses, and as discussed in Section III C, it is also very beneficial with respect to rotor lift capability but detrimental to horsepower required and rotor aerodynamic efficiency.

Additional variables affecting vibratory stress are shown in figure 34. The effects of propulsive force required and gross weight, shown in the top half of the figure, indicate that gross weight (rotor lift) is the more important of the two over the expected range of values. The lower half of figure 34 indicates that forward speed is another important variable with regard to blade stresses, as noted earlier, with stress amplitudes increasing quite rapidly with speed.

On the basis of the above considerations and those described in Sections III A and C, the rotor selected for the high performance helicopter study has the following characteristics: five blades of -4 degrees linear twist (aerodynamic twist from center of rotation to blade tip), chord of 1.52 feet (standard S-61 blade section) and a radius of 28 feet. This rotor is satisfactory with respect to both performance and vibratory stress for all of the flight conditions required to satisfy the specified performance. The calculated maximum vibratory stress amplitudes for this rotor at 175 knots and a tip speed of 650 feet per second are shown in figure 35 as a function of rotor propulsive force and lift. For the rotor propulsive force required of 1185 pounds (effective external parasite area 11.4 square feet) the assumed limit of 4000 psi vibratory stress is reached at a gross weight of 11, 800 pounds, as indicated by the open circular symbol in figure 35. This lift is slightly above the calculated stall limit, so that blade stresses are satisfactory at this tip speed at all weights up to the stall limit. For higher gross weights the tip speed must be increased to keep the stresses to acceptable values and to avoid stall. For a tip speed of 700 feet per second and a gross weight of 14,000 pounds, (the approximate stall limit at 175 knots) the calculated flatwise vibratory stress is 3, 800 psi, as indicated by the shaded square in figure 35. Thus it is not anticipated that any stress problem will be encountered at 175 knots flight speed at any of the gross weights considered. The triangular symbol on the 11,000 pound lift line of figure 35 corresponds to the operating condition of modification (c) of the high performance helicopter and is discussed in Section V C.

Check calculations made of edgewise vibratory stresses, using conventional blade stress analysis techniques with edgewise load distributions calculated by the numerical integration rotor performance method and with lag damper loads included, have indicated that the edgewise stresses are lower than the flatwise stresses, as expected. A sample result is shown in figure 36. Because of the shape of the spar cross section, edgewise and flatwise stresses can combine at the top and bottom rear corners of the spar, so that the critical stress is in reality the sum or difference of the two curves shown. With the worst possible phasing, the combined vibratory stress would be the sum of the flatwise and edgewise components, or about 5000 psi for the case shown. The actual combined stress is generally lower, less than 4500 psi in this case. It appears that the assumed limit of 4000 psi flatwise stress, based on approximately two-thirds of 5900 psi for a 500 hour life, is a conservative figure, and blade life should be substantially greater than 500 hours at the high speed condition. For a maximum combined vibratory stress of 5000 psi, blade life (derived from figure 29) would be several thousand hours.

A check was also made on this rotor to determine whether the ferry mission, with higher gross weights, might involve excessive blade stresses. At a flight speed of 100 knots and a gross weight of 20,000 pounds maximum flatwise vibratory stress amplitude was only 2200 psi. Therefore, no stress problem is anticipated at any condition at which the high performance helicopter will operate.

E. FLYING QUALITIES STUDY

1. Introduction

The stability and control characteristics of the high speed helicopter represents an additional key element in the present investigation. The flying qualities study, along with the achievement of low parasite drag and satisfactory blade stress and vibration characteristics, is a major item on which the probable success of the vehicle is based. At the same time, the flying qualities should be considered an important part of a research program to be conducted with the high performance helicopter, particularly in the modified configurations discussed in Section V. Thus, while this section is devoted primarily to achievement of satisfactory flying qualities for the selected helicopter design, it should be noted that certain features of the design, particularly the elevator provided for longitudinal control, have not been defined in detail because of the assumption that various arrangements would be tried during the flight research investigation.

The tail surfaces were selected in accordance with results of previous studies of high speed rotary wing vehicles. The vertical fin and rudder were selected to provide positive directional stability and directional control independent of the tail rotor, which may, as discussed in Section III C, lose some of its thrust capability at high speeds. This is expected to be particularly true for some of the higher speed modifications discussed in Section V, for which it may be necessary to disengage the normal tail rotor pitch control. The horizontal tail was selected to provide the longitudinal dynamic stability characteristics required, as discussed below. While the investigation shows that the use of this elevator is not required to meet the specified flying qualities, it is believed that a highly desirable stick force gradient system similar to that on a fixed wing aircraft can be provided with the elevator control, and that this development should be included in the research program. The elevator, furthermore, will provide adequate longitudinal control for later configurations where the main rotor may not be suitable for this function.

2. Background of Flying Qualities Study

Considerable concern has been expressed about the flying qualities of high speed helicopters. It has been thought that deterioration of rotor control and damping in pitch and roll would occur with high advance ratio. Furthermore, with the onset of compressibility effects and stall it was believed that cyclic pitch would not be effective in controlling rotor resultant magnitude and direction. In maneuvering loads studies for helicopters in high speed symmetrical pullouts some of these undesirable effects have been observed. An example is shown for the S-58 helicopter in figure 37. In this structural integrity demonstration performed at

Patuxent River, Md., the helicopter entered a symmetrical pullout at 126 knots. The longitudinal cyclic control was pulled aft 1.8 degrees (.03 radian) and then rapidly returned to a point 3.6 degrees (.06 radian) forward of the initial trim point. Simultaneously, collective pitch was initially increased to increase the magnitude of the normal load factor; then, with application of forward cyclic control, collective pitch was reduced. In the pitching acceleration trace it may be seen that nose-up pitching acceleration during the pullup phase was greater than the nose-down acceleration during the recovery phase even though twice as much cyclic pitch control was used for recovery than for the pullup. In the normal load factor trace the resulting load factor increases in the initial pullup and then, because of the high rate of nose-up pitch achieved during the pullup and the deterioration of nose-down pitching control moment previously mentioned, continues to build up during the recovery. This is considered highly undesirable. A reduction in the build up of nose-up rate in such a maneuver can be accomplished by increasing the angle of attack stability with a large horizontal tail. The deterioration in control moment presents a more serious problem, so study was initiated in this area.

Some of the cyclic control deterioration can be attributed to reduction in length of the rotor resultant force vector (reduction of lift) due to the decreased rotor angle of attack accomplished by application of forward cyclic control. However, this effect was not sufficient to accomplish the magnitude of control deterioration experienced; a change in the direction of the resultant force vector must also be assumed. The effect of cyclic pitch control on rotor propulsive force was therefore investigated. In figure 38 the variation of propulsive force with angle of attack for three collective pitch values is shown for a rotor at 175 knots forward speed. These curves were obtained from the numerical integration performance method discussed in Section III C. A line of constant rotor lift of 11,000 pounds is also shown. In this figure it may be seen that the slope of propulsive force with angle of attack at constant collective pitch varies with propulsive force required. For low values of propulsive force required (such as that corresponding to $\theta_{.75R} = 8$ degrees) the reduction in rotor angle of attack due to forward cyclic pitch produces, at constant collective pitch, an increase in propulsive force. As the propulsive force required increases this control derivative becomes zero and then reverses. This latter condition, which exists for $\theta_{.75R} = 12$ or 16 degrees for a lift of 11,000 pounds, results in a decrease of rotor propulsive force at constant collective pitch for a decrease in rotor angle of attack (forward cyclic pitch). This characteristic can be termed a control reversal. A diagram of what happens to the rotor resultant force at high and low propulsive force conditions is shown in figure 39. It is apparent that in the high speed case forward cyclic pitch would not only provide reduction in propulsive force but might provide a nose up pitching moment about the center of gravity, another type of control reversal. This deterioration of pitch control appears to be an advanced stage of that observed in the symmetrical pullout of the S-58 helicopter. The significance of such a control reversal is shown in figure 40 in terms of the variation of air speed following a forward cyclic pitch control change of .01 radian. The calculated time history for the S-58 helicopter at an initial speed of 90 knots is shown with the normal propulsive force-cyclic pitch control derivative and with this derivative assumed reserved. As might be expected, completely reversed air speed response to cyclic pitch control is experienced with the reversed derivative and the aircraft would initially slow down rather than speed up as desired.

An important question arose in the study at this time relative to the validity of a linear stability derivative analysis of this problem. A study of figure 38 reveals that a rapid change of the propulsive force derivative is incurred with rotor angle of attack at constant collective pitch. It would appear that an increase in rotor angle of attack occuring in a pullup maneuver could rapidly bring the propulsive force derivatives to normal values whereas a decrease in rotor angle of attack could aggravate this control reversal. A serious doubt was thus cast on the linear stability derivative analysis.

In Reference (7), NACA investigators predicted that high speed helicopters could incur unstable rotor damping in pitch and roll if the ratio of collective pitch to thrust coefficient-solidity ratio were on the order of 3 or higher. These investigators believed that the helicopter would be difficult to control under these conditions unless cyclic pitch rate stabilization were incorporated. However, this study was performed with simplified rotor theory which is not considered accurate for high advance ratios and inflow angles.

The foregoing considerations and a number of other problems helped to determine the need for a refined flying qualities analysis which could be set up on an analog simulator and which would incorporate the effects of compressibility, stall, high advance ratios and inflow angles, and not depend on the assumption of linear stability derivatives. It is believed that without such a refined theory no reasonable guarantees could be made concerning the compliance of the design study aircraft with military specifications for flying qualities.

3. Refined Flying Qualities Design Analysis

A refined analysis has been developed for the study of flying qualities of high performance helicopters. In this analysis essentially the same equations of rotor flapping and feathering behavior are solved as in the numerical integration performance method discussed in Section III C. However, a large scale analog computer is used instead of a digital computer and solutions are performed in real time. A brief description of the method is as follows:

a. Airfoil section data as a function of angle of attack and Mach number are set up in function generators to be used in the aerodynamic force and moment integrations.

b. Uniform inflow and rigid blades are assumed.

c. Actual shears are calculated at the flapping hinge in order that hub moments due to the presence of offset flapping hinge may be accurately calculated.

d. Complete equations of motion of the helicopter body in 6 degrees of freedom are solved. These are non-linear total force equations with no simplifying small angle assumptions.

e. Wind tunnel data obtained from a 1/10th scale model of a high speed helicopter including lift, drag, pitching moment, side force, and yawing moment are incorporated.

f. The analysis is controlled by means of a fixed platform simulator shown in figure 41. The simulator consists of complete helicopter flight controls and an optical instrument display. The instrument display consists of an oscilloscope displaying roll, pitch, yaw, rate of climb, and sideslip information and electric indicators for air speed, altitude, control positions, and rotor horsepower.

To summarize, this analysis is of such character as to permit "flying" the helicopter from hovering to maximum speed and back, including accelerated flight conditions as well as static trim. The effects of compressibility, blade stall, large inflow angles and high advance ratios are automatically included.

4. Calculated Flying Qualities

a. General Description of Study Because of time restrictions, it was necessary to base the flying qualities study on a preliminary version of the high performance helicopter. The characteristics of the helicopter simulated vary slightly from the final design with respect to rotor solidity and a few other details but these differences are not believed to be important in the flying qualities evaluation. The physical characteristics of the simulated vehicle are presented in Table 2 and the aerodynamic characteristics of the body, obtained from tests of a 1/10th scale model in the UAC Pilot Wind Tunnel, are shown in figure 42.

The flying qualities of the high speed helicopter were determined by "flying" the simulator through flight test programs similar to those that would actually be flown in a real aircraft. In those areas considered most important,a comparison is made of the calculated flying qualities with those proposed in Reference (8) (MIL H 8501-A). This Mil Spec contains all of the specifications of MIL H 8501 but is used instead of the latter because it is more comprehensive.

The helicopter was studied at two gross weights, 11,000 pounds, corresponding to the high speed flight condition, and 17,000 pounds, corresponding to a more moderate speed mission or to the ferry mission. Three center of gravity positions were considered, covering a total range of 15 inches: a normal c.g. directly beneath the rotor head, a forward c.g. 7.7 inches forward of normal, and an aft c.g. 7.3 inches aft of normal. All conditions were flown for a sea level standard day.

b. Longitudinal Static Trim Static trim parameters for level flight, including cyclic and collective pitch values, fuselage pitch attitude, and the rotor hub pitching moment are presented in figure 43 and 44. Results obtained for the normal c.g. position with a digital computer analysis containing conventional Bailey rotor theory are also shown for reference. It is of importance to note that the refined theory calculates 4 degrees more longitudinal control and 1 degree more collective pitch control required at 175 knots (figure 43) than the standard theory, and it the control system were designed on the basis of the standard theory, there would probably be too small a pitch angle range available to permit flight at the maximum design speed.

Figures 43 and 44 indicate that the helicopter possesses positive longitudinal control position stability with respect to speed as required by MIL H 8501 for both gross weights considered. If figure 45 is examined, however, a specific requirement of paragraph 3.2.10 of Reference (8), regarding cyclic control required to hold increased and decreased airspeeds at constant collective pitch, is satisfied at 104 knots but is not satisfied at 176 knots since, for small deviations from the trim speed, no change in cyclic pitch is required. An attempt to remedy this lack of speed stability by reducing tail incidence 5 degrees was not entirely successful and increased the forward longitudinal control requirements considerably. It is believed that this condition, which might be mildly objectionable, can best be corrected by utilizing the horizontal tail surfaces and introducing a control force gradient developed by a tab control surface on the horizontal stabilizer. This should be considered part of the flight research program to be conducted with the vehicle.

c. Longitudinal Maneuvering Characteristics The helicopter was "flown" to a top speed of 200 knots without the use of auto-pilots. No appreciable deterioration in control response was experienced even though the rotor was stalled at this speed. Damping in pitch appeared entirely satisfactory. Only the speed stability was undesirable at high speeds. The aircraft exhibited a long period pure divergence producing either climbs or descents characteristic of this lack of speed stability. This condition was aggravated by the aft c.g. loading whereas the forward c.g. loading condition exhibited greatly improved speed stability.

To determine the sensitivity of the flying qualities to aerodynamic cleanliness, the parasite drag was doubled on the simulator. In this way the ratio of collective pitch to thrust coefficient-solidity ratio was increased from 3.7 to 4.6 and, according to Reference (7), should have produced unstable rotor damping in pitch. No appreciable deterioration in flying qualities occurred even though the rotor was stalled at 175 knots. Furthermore, this increase in parasite drag did not produce a pitching control reversal though there was marked control power deterioration. It is emphasized that the presence of a large horizontal tail contributing angle of attack stability and damping in pitch greatly improves the flying qualities of this aircraft.

To determine the effect on flying qualities of the offset flapping hinge, the offset was "removed" from the simulated rotor to reduce the longitudinal control power of the helicopter. The helicopter was flown at normal parasite drag to a speed of 175 knots. A noticeable control reversal was experienced on application of forward longitudinal cyclic control. A time history of this situation is shown in figure 46. The aircraft under these circumstances would not comply with paragraph 3.2.9 of Reference (8). It is apparent that an offset flapping hinge is very desirable in a high speed helicopter to prevent the deterioration of longitudinal pitching control.

Time histories of longitudinal control response with normal flapping hinge offset are shown in figures 47 and 48 for a gross weight of 11,000 pounds at 175 knots, and 17,000 pounds at 150 knots, respectively. Maneuvering responses at these conditions are believed to be the most critical of any flight conditions that will be encountered. Although a maximum load factor of only 1.3 was obtained in the pull-and-hold maneuver represented in figures 47 and 48, a full inch of longitudinal cyclic control was used in each case. It is believed that these time histories have the same characteristics as those developed in a maneuver in which a load factor of 1.5 would be developed. (See paragraph 3.2.11.1 of Reference (8).) Compliance with paragraphs 3.2.11.1(a) and (b), and 3.2.12 of Reference (8) with regards to the shape of load factor and pitching rate curves is easily attained at these maximum speeds for each gross weight. Likewise, compliance with these paragraphs is also attained at lower speeds as might be expected of a conventional single rotor helicopter with offset flapping hinges fitted with a large horizontal tail.

Time histories of longitudinal cyclic control-induced disturbances for gross weights of 11,000 pounds at 175 knots, and 17,000 pounds at 150 knots are shown in figures 49 and 50 respectively. In compliance with paragraph 3.2.11.2 of Reference (8), a load factor change of less than 0.25 from trim is experienced within either 10 seconds of the disturbances or 10 seconds after passing through the initial trim attitude on the nose down swing.

d. Longitudinal Control Power Since the helicopter studied possessed typical hovering response characteristics no special emphasis was placed on a study of hovering flying qualities. Control power and damping in pitch in hovering were checked with required characteristics set forth in paragraphs 3. 2. 13 and 3. 2. 14 of Reference (8). In accordance with paragraph 3. 2. 13 of Reference (8), a 1 inch longitudinal control input should produce at least 1. 7 degrees of pitch attitude change after 1 second. According to the simulation 9 degrees of pitch attitude change are attained after 1 second which also more than satisfies the requirement for a pitch change of at least 6. 8 degrees after 1 second using maximum available control motion. In paragraph 3. 2. 14 of Reference (8) a damping moment for this aircraft of at least 9700 foot pounds per radian per second is required. The damping moment calculated by the simulation was 23, 000 foot pounds per radian per second, more than meeting the requirement.

The above characteristics may be attributed mainly to two design details: (1) offset flapping hinge with large hub moments coming from the 5 bladed rotor as a function of both control displacement and angular rate; and (2) a large amount of cyclic control per inch of longitudinal stick displacement. The latter is determined by the static trim requirements, shown in figures 43 and 44, of a large amount of longitudinal cyclic pitch in high speed flight. In flying the simulator there is no indication that the controls are unduly sensitive, in spite of the high control power exhibited by the configuration in low speed flight. This can be attributed to the high damping in pitch also present.

e. Longitudinal Dynamic Stability An examination of figures 49 and 50 reveals that the aircraft falls into category (a) of paragraph 3. 2. 11 of Reference (8) at the maximum speed for each gross weight. An oscillation possessing a period of about 3 seconds is damped to an insignificant level in 1 cycle. At lower forward flight speeds the aircraft moves into category (c) and complies with paragraph 3. 2. 11 because it possesses dynamic stability above 50 knots airspeed.

f. Lateral Static Trim As typical of a single main rotor helicopter, the aircraft possesses positive control-fixed directional stability and effective dihedral above 50 knots airspeed. These characteristics are in compliance with paragraph 3. 3. 9 of Reference (8). Curves of static trim as a function of slip angle are shown in figures 51 and 52 for gross weights of 11, 000 pounds at 100 and 175 knots, and 17, 000 pounds at 100 and 150 knots, respectively.

g. Lateral Maneuvering Characteristics With the exception of coordinated turns, no unusual lateral maneuvering characteristics developed in the simulator study. In coordinated turns at constant speed it was observed that whereas additional collective pitch is required to maintain constant altitude at ordinary flying speeds, a slight reduction in collective pitch was required to maintain constant altitude at 175 knots. This phenomenon may be observed in the chart presented in figure 53 in which rotor thrust versus collective pitch is plotted for a variety of parasite drag requirements for a typical helicopter rotor flying at 175 knots. For a constant altitude coordinated banked turn, an increase in thrust is required relative to steady level flight, and it is noted that for constant parasite drag a decrease in collective pitch is required to cause an increase in thrust over much of the area of the chart. This "control reversal" is minimized for increasing thrust requirement and decreasing propulsive force or parasite drag requirement. A graph of collective pitch required as a function of bank angle for two airspeeds and parasite drag levels is shown in figure 54. At 100 knots there is the conventional increase in collective pitch for increased angle of bank. At 175 knots, however, the collective pitch gradient is negative up to 60 degrees of bank at a parasite area of 18 square feet. For a parasite area of 13 square feet, the collective pitch gradient is negative up to 35 degrees of bank and positive at higher bank angles. In the simulator studies this characteristic was not considered annoying, and the technique of achieving properly coordinated turns was accomplished without difficulty.

F. AIRCRAFT DESIGN DETAILS

1. Fuselage

The fuselage is of conventional semi-monocoque construction and similar to others that Sikorsky has built for this general weight class of helicopter. For this reason, no special emphasis on this phase of the preliminary design is believed to be required. However, during the detail design phases of the high performance helicopter, the fuselage natural frequencies and responses to the expected forcing functions from the main rotor would be investigated to ensure that fuselage vibration characteristics would be satisfactory. Because of the relatively continuous load paths provided by the basic structural layout, no special problems are anticipated.

As shown in figure 8 an access door is provided on the left side of the fuselage just behind the cockpit, opposite the controls "closet" on the right hand side of the ship. This would be an "air-stair" door, and would be the only access required for a research investigation. However, provision for a larger cargo door would be provided on the right side of the fuselage in the center or rear of the cabin area.

The cockpit is essentially identical to the S-62 helicopter cockpit, except that structural reinforcement is required because of the higher aerodynamic loads on the canopy, and a new external shell is provided on the bottom (below waterline 120, figure 8).

The tanks for the normal fuel load (1700 pounds) are located below the cabin floor. For longer ranges, space is available for an additional tank below the floor immediately aft of the main tank. For the ferry mission it is assumed that the auxiliary tank will consist of a large rubber bag in the cabin immediately above the normal main tank. This technique has been successfully applied in a previous helicopter ferry mission application, wherein the bag was suspended from above, with a rigid rim around the top of the bag supporting the tension in the bag material. Figure 55 shows a tank of this type which was installed in an S-58 helicopter. By providing fittings on the cabin ceiling, provision for quick installation of this type of ferry tank is easily obtained without the necessity of auxiliary structure. The total weight for the ferry tank system, which would feed fuel by gravity to the main tanks, is estimated to be not more than 200 pounds.

2. Landing Gear

The landing gear is fully retractable with doors to seal the openings in the bottom of the fuselage when the gear is up. The main gear has a tread of 10 feet, and is basically similar to the latest production gear (non-retractable) used on the S-58 helicopter. This type of gear, with the wheel pivoting about a longitudinal axis at the bottom of the aircraft, has been shown to have superior characteristics with respect to ground resonance problems. Ground resonance for this type of helicopter is characterized by a rolling oscillation of the fuselage when rotor lift is nearly equal to gross weight and the gear is partially extended and in light contact with the ground. With the recommended gear system, a rolling motion in this condition will cause a maximum displacement of each wheel, so that energy absorption by the shock struts is maximized. Another important factor is that the shock struts are doubly pin-ended, so that there will be no bending moment tending to cause the struts to bind.

Tires are 18 inch diameter, high pressure type (18 x 5.5, type 7). While not suitable for landing on soft ground, ordinary unprepared hard ground will be a perfectly acceptable surface for vertical take-off and landing operations.

Retraction is accomplished by breaking the forward (drag) strut in the approximate center and rotating the gear forward and upward about a laterally-inclined hinge axis passing through the inboard ends of the rear main strut and the shock strut. Tail wheel retraction is accomplished by simple forward rotation of the entire unit. In case of hydraulic system failure, it will be possible to extend both main and tail gear by the combination of gravity and aerodynamic drag.

3. Rotor Fairing System

The rotor fairing system will be essentially identical to the test model described in Section III B except for materials and fabrication techniques. The boundary layer control air would be supplied most efficiently by a multi-stage axial flow fan geared to the tail rotor shaft. Analysis by blower manufacturers has indicated that a unit with a diameter on the order of 10 inches and a weight of about 20 pounds would be adequate. Since the mass flow requirement for the boundary layer control system is of the same order of magnitude required for transmission oil cooling, there is a good possibility that these two requirements could be combined effectively in a single system. This would eliminate the need for the normal oil cooler air exhaust port and reduce auxiliary inlet air requirements.

4. Engine Inlet and Exhaust

Engine inlet design is changed relative to the S-61 by tailoring it to more efficient operation at high speeds. While the present bellmouth inlets are completely satisfactory as far as internal flow is concerned, the inlet area is excessive for the high speed flight condition considered here, and excessive external drag would arise because of the resultant flow spillage. Accordingly, the inlet has been reduced in area and moved forward to permit a more streamline external shape. The new cowling would be made of fiberglas-plastic to provide the required shape at minimum cost.

Engine exhaust, as discussed in Section III C, is extended slightly and equipped with a row of curved deflection vanes to redirect the exhaust to 30 degrees from the longitudinal axis. This will be accomplished without a decrease in exhaust area, so that no back pressure will be developed on the engine.

5. <u>Controls</u>

Flight controls are a combination of existing S-61 and S-62 hardware. Cockpit controls are identical to S-62 components, and the two systems are mated in the controls "closet" behind the cockpit. Except for changes in length of some of the push-pull rods and relocation of control horn brackets, the only additional change required is a modification of the longitudinal control primary servo mounted on the main transmission. The stroke of this servo must be increased to provide the increased longitudinal cyclic pitch control required for high speed flight, as discussed in Section III E.

G. TRANSMISSION AND ROTOR SYSTEM STUDY

1. Main Transmission

The present S-61 main transmission designed for the T-58-8 turbine engines has normal input and output rpm values of 18,966 and 203, respectively, providing a tip speed of 660 feet per second with a 31 foot radius rotor. Rpm values can be increased or decreased over a small range from this design point without loss of engine power. For the present high performance helicopter design, the rpm requirement is above that obtainable with the standard transmission. The gear ratio can be changed by a simple substitution of gears in the second stage of reduction. The number of teeth on two gears (one for each engine) would be increased from 43 to 47, and on the single gear that couples the two power units together, the number of teeth would be reduced from 109 to 105. These are the only changes required in the main gearbox; all present housings, bearings, etc. would be unaffected. This gearing change increases the output rpm by 13.5 percent, providing a normal output of 230 rpm and a tip speed of 675 feet per second with a 28 foot radius rotor. The desired tip speed range of 650 to 700 feet per second may then be obtained with full engine power available over this range. Because the present transmission is torque-limited with respect to power capacity, the increased rpm results in a corresponding higher maximum horsepower capacity (increased from 2100 horsepower maximum to approximately 2400 horsepower). At constant torque, however, the life in hours will be reduced by the same percentage because of the increased cycles per hour. On the other hand, for a given horsepower transmitted, transmission life may be expected to increase because of reduced torques.

2. Tail Rotor Drive Shaft

A universal joint to droop the tail rotor drive shaft 3 degrees from its present position on the S-61 is added aft of the main transmission oil cooler. Drive shaft and bearings are the same, except for bearing spacing. Disconnect coupling for tail pylon folding on HSS-2 is not required.

3. Intermediate Transmission

This gearbox requires modification from the present unit because of a change in direction of the output shaft and because of the change in gear ratio of the main transmission. To maintain the present tail rotor rpm, the gear ratio would have to be changed from 1 : 1 to 1 : 1.135 (or approximately so) because of the increased input shaft speed. The number of teeth on both gears would be changed as well as the included angle between input and output shafts. The middle section of the present three-piece housing would be changed, but the input and ouput housings and bearings should be satisfactory without change.

4. Tail Gear Box

No changes are required.

5. Rotor System

The rotor head is the same as the S-61 commercial version except that the droop stops (to prevent negative blade flapping angles when the rotor is stopped on the ground) would require slight modification to avoid interference with the cylinder installed below the rotor head as part of the rotor head fairing system. The stationary scissors on the swash plate would also require modification for the same reason.

The main rotor blades differ from the present production blades only in length and twist. The reduced length is obtained by removing three additional feet from the spar tip during manufacture. Because twist is normally applied after the spar machining operation, obtaining a new desired twist involves no additional operations. All other features of the blade remain the same except for a non-structural fairing applied behind the spar inboard of the root-end trailing-edge pocket.

Tail rotor head and blades are unchanged from the present design.

H. GROUP EMPTY WEIGHT STATEMENT

Main Rotor Group		2113.1 lb.
Blades	905.0	
Head and Controls	994.1	
Fairing and Boundary Layer Control	214.0	
Tail Group		344.2
Tail Rotor Blades	30.5	
Rotor Head System and Fairing	86.7	
Horizontal Tail	100.0	
Pylon (fin, rudder & controls)	127.0	
Body Group		1548.2
Landing Gear Group		345.0
Engine Section		116.0
Power Plant Group		2396.8
Engines (GE-T58-8)	532.0	
Engine Accessories	18.7	
Power Plant Controls	13.2	
Gear Boxes	1386.1	
Shafting	111.0	
Rotor Brake & Controls	40.6	
Transmission Oil Cooler	58.0	
Starting System	32.5	
Lubricating System	42.5	
Fuel System	162.2	
Fixed Equipment		1161.7
Instruments	134.0	
Flight Controls	305.6	
Hydraulic System	95.1	
Electrical System	383.0	
Electronic (radio)	50.0	
Furnishings	194.0	
WEIGHT EMPTY		8025.0 lb.

A study of the center of gravity for anticipated loadings at all gross weights indicated that the center of gravity range is within the limits considered in the flying qualities study (Section III E).

SECTION IV: SPECIFICATION AND PERFORMANCE SUMMARY

A. DIMENSIONAL DATA

1. Rotor System

	Main Rotor	Tail Rotor
Diameter	56 ft.	10 ft.
Number of Blades	5	5
Chord	18.25 in.	7.35 in.
Solidity	.086	.195
Twist	-4 deg.	0 deg.
Airfoil Section	0012	0012
Flapping Hinge Offset	12.625 in.	3.625 in.

2. Fuselage

Length	54 ft.
Width	70 in.
Height	9 ft. 3 in.
Cabin Length	20 ft. 3 in.
Cabin Width	64 in.
Cabin Height	60 in.

3. Empennage

Horizontal Tail

Area	50 $ft.^2$
Span	15 ft.
Aspect Ratio	4.5
Airfoil Section	0015
Elevator Area	15 $ft.^2$

Vertical Tail

Area	40 $ft.^2$
Height	9 ft.
Aspect Ratio	2.0
Airfoil Section	0020
Rudder Area	10 $ft.^2$

4. Overall Dimensions

Length (Rotors Turning)	66ft. 5 in.
Height (Ground Static Line to Top of Rotor Fairing)	13 ft.
Width (Rotor Turning)	56 ft.
(Main Blades Removed)	15 ft.
Landing Gear Tread	10 ft.

B. WEIGHTS

	Normal Mission 175 Knots	150 Knots	Ferry Mission
Weight Empty* (includes 50 lb. Radio)	8025	8025	8025 lb.
Oil & Trapped Liquids	115	115	115
Pilot(s)	200 (1)	200 (1)	400 (2)
Reserve Fuel	200	200	360 (1 hr. at 10,000 ft.)
Ferry Tank	---	---	200
Operating Weight Empty	8540	8540	9100
Fuel and Payload*	5060	8460	8900
Gross Weight	13,600	17,000	18,000 lb.

* For fuel weights above 1700 lb. in normal mission, 5 percent of additional fuel weight should be allowed for increased tank weight.

C. POWER PLANT

Two General Electric T-58-8 free power turbine engines. Rated power each engine (sea level static, standard day):

Military (30 minutes)	1250 brake horsepower
Normal (continuous)	1050 brake horsepower

Specific fuel consumption at normal rated power 0.634 pounds per horsepower hour.

D. PERFORMANCE

Top Speed at Minimum Weight (Sea Level)	195 knots
Top Speed at 13,600 pounds (Sea Level)	182 knots
Max. Cruise Speed at 13,600 pounds (Sea Level)	175 knots
Cruise Speed at 17,000 pounds (Sea Level)	150 knots
Max. Rate of Climb at Sea Level at 13,600 pounds (Normal Rated Power)	3000 ft/min.
Max. Rate of Climb at Sea Level at 17,000 pounds (Normal Rated Power)	1900 ft/min.
Hovering Ceiling Out of Ground Effect at 13,600 pounds (Military Power)	14,200 ft.
Hovering Ceiling Out of Ground Effect at 17,000 pounds (Military Power)	6500 ft.
Single Engine Hovering Ceiling in Ground Effect at 13,600 pounds (Military Power)	1100 ft.
Ferry Range at 18,000 pounds Gross Weight (Vertical Takeoff, Single Engine Cruise)	2080 n. miles +1 hour fuel reserve
Range with Normal Fuel (1500 lb. + 200 lb. Reserve) at Cruise Speed	200 n. miles + reserve

(Above figures are all for standard day operation)

Payload range characteristics for the high performance helicopter are shown in figure 56 for three flight conditions: 175 knots cruise at sea level, 150 knot cruise at sea level, and the ferry mission (single engine cruise at variable speed and altitude). Also shown for reference are the payload-range curves for two helicopters of comparable size: the S-58 helicopter at a normal gross weight of 13,000 pounds and a cruise speed of 87 knots, and the S-61 helicopter at a normal gross weight of 17,300 pounds cruising at 130 knots. As may be seen, the high performance helicopter has a payload-range curve at 175 knots similar to the S-58 cruising at approximately one half of this speed. At 150 knots, the high performance helicopter has payload-range characteristics superior to the S-61 at 130 knots, partly because of the greatly reduced parasite drag, and partly due to elimination of several empty weight items on the S-61 (flying boat hull, automatic blade folding, etc.)

Ferry range of the high performance helicopter is considerably greater than for the two reference helicopters. The ferry range of the S-58 at a gross weight of 14,000 pounds is calculated to be 1010 nautical miles, and the ferry range of the S-61 at 19,000 pounds is 1290 nautical miles. The ferry range of the high performance helicopter (2080 nautical miles at 18,000 pounds) is at comparable loading conditions for the rotor size and installed power. This range represents increases of more than 100 percent and 60 percent over the ferry ranges of the S-58 and S-61, respectively.

Additional performance data are presented in Section III C (figures 14-23).

E. PRODUCTIVITY STUDY

To illustrate the benefits of increased speed and range potential on the capability of performing useful missions, a brief study of aircraft productivity is presented. In figure 57 the effect of flight speed on block speed is illustrated. Block speed, defined as block distance divided by the sum of flight time and turn-around time, is shown as a function of block distance and turn-around time for flight speeds of 100 and 175 knots. This chart illustrates that for the shortest distance considered (10 miles), turn-around time is much more important than flight speed; it is at short ranges, therefore, that the "flying crane" helicopter, with moderate speed but with extremely rapid loading and unloading features, has its greatest potential. On the other hand, for ranges more than 50 nautical miles, increased speed capability provides substantial benefits, even if a penalty in turn-around time is accepted. It is for the longer ranges (50 nautical miles or more) that the high performance helicopter will have its greatest advantage in block speed over slower machines.

The productivity, or rate of accomplishing a useful transfer of material, is defined as the product of the payload and the block speed. The productivity of the high performance helicopter cruising at 150 and 175 knots, and comparison with the S-58 and S-61 at typical cruise speeds, is shown in figures 58 and 59 for turn-around times of 5 and 20 minutes, respectively. Because of the increased payload permitted, the productivity of the high performance helicopter is substantially greater at 150 knots than at 175 knots, and thus the lower speed represents a more practical cruise for most missions. The high performance helicopter at 150 knots also demonstrates considerably better productivity than the S-61 at 130 knots, and far greater productivity than the S-58 at 87 knots. The high performance machine, therefore, represents not only a useful research tool for high speed helicopter flight, but provides a considerable step forward with respect to capability of transferring passengers or cargo over useful distances.

SECTION V: CONFIGURATION MODIFICATIONS FOR ADDITIONAL PERFORMANCE IMPROVEMENTS

In addition to the basic high performance helicopter, various configuration modifications were studied briefly to determine possible benefits to speed, payload, or range. It was found that none of the particular modifications considered offered any significant increase in range capability, but substantial increases in speed and payload were found to be possible. The four modifications considered were (a) pure helicopter with a sixth blade added to the main rotor (20 percent increase in rotor solidity); (b) six blades plus a turbojet engine for auxiliary lift and propulsion; (c) basic 5 blade helicopter plus a wing for auxiliary lift; and (d) basic 5 blade helicopter plus a wing and two turbojet engines (jet compound). It should be noted that all of these modifications incorporate increased lift capability over the basic high performance helicopter, because of the rapid drop of rotor lift capability with increasing speed in the speed range under consideration.

A. MODIFICATION (a): PURE HELICOPTER WITH INCREASED ROTOR SOLIDITY

While the basic high performance helicopter meets the design payload requirements at 175 knots, it is important to note that the rotor lift capability at this speed is considerably less than at lower forward speeds or in hovering. The maximum cruise gross weight of the basic 5-blade HPH was established as 13,600 pounds at 175 knots compared to 17,000 pounds at 150 knots and 18,000 pounds in hovering. A logical way to circumvent part of this loss in lifting capacity at high speeds is to increase main rotor solidity, either by increasing blade chord or the number of blades. The simplest procedure in the present case is to add a sixth blade, because Sikorsky Aircraft has already designed and built a six-bladed S-61 rotor head. Except for the increased weight of the rotor system (380 pounds) and the new rotor head fairing required to accommodate six blades instead of five, the six-bladed helicopter is in all other respects identical to the basic HPH.

It might be expected that the increase in rotor solidity of 20 percent would provide an increase in lifting capacity of 20 percent. While this would be true if the installed power were unlimited, in the present case less than 20 percent is achieved. The basic HPH operates at the normal power limit of the engines at a speed of 175 knots and a gross weight of 13,600 pounds so that the profile power of an additional blade at the same operating conditions would raise the total power above the available limit. To reduce the profile power of the six-blade version, the tip speed of the rotor is reduced from 700 to 675 feet per second. At this latter tip speed, a gross weight of

15,400 pounds can be flown at 175 knots, within the normal rated power limit of the engines. This represents a weight increase of 13 percent over the basic HPH.

A comparison of the basic five-blade HPH and the six blade version at 175 knots is shown in the following table.

Configuration	Basic HPH	6-blade HPH (Modification (a))
Number of blades	5	6
Design Cruise Speed, kt.	175	175
Tip Speed, ft./sec.	700	675
Parasite Area, $ft.^2$	13	13
Rotor System Horsepower	2230	2230
Weight, lb.		
Basic Weight Empty	8025	8405
Pilot	200	200
Trapped Fluids	115	115
Reserve Fuel	200	200
Operating Weight Empty	8540	8920
Fuel + Payload	5060	6480
Gross Weight	13,600	15,400
Payload for 100 nautical mile range, lb.	4300	5600

While the increased gross weight afforded by the extra blade is only 13 percent, the increase in payload and productivity for 100 nautical miles range is approximately 30 percent. The payload-range curve for this modification as well as those for the basic HPH at 175 knots and 150 knots are shown in figure 60, and a similar comparison of productivity is shown in figure 61. It is of interest to note that while a considerable gain is afforded by the extra blade at 175 knots, the productivity still falls considerably short of the basic HPH cruising at 150 knots.

The maximum high speed capability with zero payload was calculated to be 195 knots, the same as for the basic HPH. Use of the extra blade requires a lower tip speed to avoid excessive profile power losses, so that advance ratio is increased, which adversely influences the lifting and propulsive force capabilities. These effects apparently cancel the benefit of the extra blade at this high speed. If more engine power were available, the six-blade HPH would have a top speed potential of at least 200 knots. Ferry range of this modification is no greater than for the basic HPH. While the higher rotor solidity would allow cruising at higher altitudes, thus decreasing parasite power, this advantage is offset by the weight penalty of the extra blade.

B. MODIFICATION (b): INCREASED ROTOR SOLIDITY PLUS A TURBOJET ENGINE FOR AUXILIARY LIFT AND PROPULSION

It has been established by various studies, such as Reference 2, that a helicopter rotor loses both lifting and propulsive force capability as forward speed is increased. While lifting capability remains positive at all speeds investigated, the propulsive force capability disappears completely at a speed dependent on rotor geometry, tip speed, and required rotor lift. While the maximum speed for zero propulsive force is considerably in excess of 200 knots for the HPH rotor, the "practical" speed limit of the pure helicopter is obviously not as high. The practical limit may be defined as the maximum speed at which the rotor can produce a sufficient propulsive force to overcome fuselage parasite drag simultaneous with generating enough lift to provide an economically useful payload. For the present pure helicopter configuration the design cruise speed of 175 knots appears to be close to the economical limit; at higher speeds the payload or power penalties become excessive.

In order to provide increased economical speeds, auxiliary propulsion appears to be required. Accordingly, the second modification of the HPH consists of the addition of a turbojet engine to provide a cruise speed (with payload) of 200 knots. A turbojet rather than a turbine-propeller combination for auxiliary thrust was selected because of considerably less cost and complexity for a research installation. Another advantage is that for relatively short ranges, the weight of additional fuel consumed by the turbojet is more than offset by the considerably lower installed weight than that of a turbine-propeller unit having the same thrust capability in cruise. This configuration, which retains the main rotor blade added in modification (a), is shown in figure 62.

The turbojet selected for the installation is the Pratt and Whitney JT-12 (J-60-YP3) having a static military thrust rating of approximately 3000 pounds for a bare engine weight of only 436 pounds. Other jet engines, including turbofan versions, were considered but the JT-12 appeared to be most nearly suited to the thrust requirements of the configuration. An equivalent turbofan installation would provide considerably improved static engine performance compared to the straight turbojet, but the static performance is unimportant for the helicopter installation since the auxiliary thrust is not required at low speeds. At the cruising speed of 200 knots, the thrust or specific fuel consumption advantage of the turbofan is not great enough to justify its extra weight and complexity for moderate ranges.

Three features of the turbojet installation should be noted. First, the engine is located on the right side of the fuselage. The purpose of this location is to provide a yawing moment in the direction to counteract main rotor torque, thereby reducing the tail rotor power (1000 pounds of turbojet thrust counteracts the torque corresponding to approximately 200 main rotor horsepower). Second, the engine is located as far to the rear as practical to eliminate possible jet exhaust interference effects on the tail surfaces over the expected angle of attack range. Third, the engine thrust is canted at an angle of 15 degrees to the aircraft longitudinal axis. The inclined thrust axis provides a significant amount of auxiliary lift from the turbojet as well as the propulsive force. This auxiliary lift is equal to the gross jet thrust (net thrust plus the inlet ram drag) times the sine of the inclination angle, which for the operating condition at 200 knots amounts to 680 pounds for an engine net thrust of 2120 pounds. Loss of horizontal thrust component is only 90 pounds corresponding to an equivalent incremental lift-drag ratio of 7.5. As this approaches the lifting efficiency of the rotor, and since rotor lift is limited by the forward speed, the engine lift available is considered highly desirable.

Rotor tip speed is reduced to 650 feet per second to avoid excessive Mach number losses at the cruise speed of 200 knots. The optimum rotor power level was not determined, but from previous studies it is believed desirable to allow the rotor to overcome its own drag at this speed, that is, to operate at a propulsive force level of zero. At these conditions, the rotor system absorbs 1600 horsepower and produces slightly less than 15,000 pounds of lift at the stall limit. The auxiliary turbojet overcomes the entire fuselage parasite drag; the normal thrust rating of the JT-12 at 200 knots is just slightly greater than required. The parasite area was assumed to be 15 square feet, 2 square feet higher than for the basic HPH to account for the inclined jet engine pod, giving a parasite drag of 2030 pounds at 200 knots at sea level.

A comparison of this modification with the basic HPH is shown in the following table.

Configuration	Basic HPH	6-blade HPH plus jet (Modification (b))
Number of blades	5	6
Design Cruise Speed, kt.	175	200
Tip Speed, ft. /sec.	700	650
Parasite Area, $ft.^2$	13	15
Parasite Area Propelled by rotor, $ft.^2$	13	0
Parasite Area Propelled by jet, $ft.^2$	----	15
Rotor Lift, lb.	13,600	14,920
Jet Reaction Lift, lb.	----	680
Rotor System Horsepower	2230	1600
Weights, lb.		
Basic Weight Empty	8025	8405
Jet Engines (Installed)	----	900
Trapped Fluids	115	115
Pilot	200	200
Reserve Fuel	200	200
Operating Wt. Empty	8540	9820
Fuel + Payload	5060	5780
Gross Weight	13,600	15,600
Payload for 100 nautical mile range, lb.	4300	4150

Payload-range and productivity curves for this modification are shown in figures 63 and 64 respectively. Note that while the payload of the helicopter plus jet is slightly inferior to the basic 175 knot HPH at 100 nautical miles range, the productivity is higher because of the higher speed. The productivity is considerably inferior, however, to that of the basic HPH at 150 knots. At ranges much above 100 nautical miles, the helicopter plus jet suffers considerably from the high fuel consumption of the jet, so this modification should be considered applicable to short ranges only. However, range could be extended by shutting down the jet engine after any selected range and continuing the flight as a pure helicopter at a speed of 175 knots or less; this technique could be useful in a mission requiring maximum speed for only a portion of the flight.

Brief consideration was given to the possibility of using jet thrust to augment rotor lift in hovering. This did not prove to be desirable for the following reasons: (a) lifting capability in hovering is already in excess of the total lifting capacity at the design speed of 200 knots; (b) variable geometry of some type, either a rotatable engine pod or a variable exhaust deflector, would be required; (c) if the jet thrust is vertical or approximately so, exhaust recirculation would limit operation to well-prepared surfaces; (d) if the thrust axis is inclined at an angle sufficient to avoid recirculation (on the order of 45 degrees), the pitching moment about the aircraft center of gravity and/or the horizontal thrust component would require excessive corrective cyclic control on the main rotor and an excessive tail-down hovering attitude.

C. MODIFICATION (c): HELICOPTER WITH WING

A logical way to supplement the lifting capacity of the rotor at high speed is to use a wing. Accordingly, the third modification of the basic high performance helicopter is the addition of a small wing of moderate aspect ratio. The configuration cannot be expected to cruise at a significantly higher speed than the pure helicopter without increasing the installed power; a design cruise speed of 175 knots was therefore chosen for this modification.

Because of the high lift-drag ratio of the wing, the propulsive force demand on the main rotor is not seriously increased over the basic HPH, and the 5-bladed rotor has adequate capability at this speed. In fact, it is desirable to reduce rotor tip speed from 700 feet per second to 650 feet per second to reduce profile power losses; even at the reduced tip speed, the rotor propulsive force capability is adequate, providing the rotor lift is reduced to 11, 000 pounds. Wing lift was set at 7000 pounds to obtain a gross weight of 18, 000 pounds, the maximum weight for which reasonable hovering performance can be provided. Wing design lift coefficient was selected at approximately 0. 6 which, while somewhat higher than the aerodynamic optimum, minimizes the size and weight of the wing. Using a 63_2A415 airfoil section and an aspect ratio of 6, a lift-drag ratio of 24 would be obtained in free air. Because of the downwash from the rotor, however, there is a slight rotor-wing interference which reduces the L/D to 23 at 175 knots. At lower forward speeds the interference would be more severe. In calculating this interference effect it was assumed that the wing operates in a downwash field equivalent to the rotor mean induced velocity in the plane of the disk. Experimental studies have indicated that this represents a conservative estimate of rotor-wing interference.

The high performance helicopter with wing, modification (c), is shown in figure 65. The wing has a span of 26 feet, an area of 110 square feet including the region blanketed by the fuselage, a taper ratio of 0. 6, and an aspect ratio of 6. 1. The wing is located directly below the main transmission and is structurally tied into the transmission support structure. The aerodynamic center of the wing is directly below the main rotor hub centerline so that the influence of the wing on aircraft stability characteristics will be minimized. Because of the small size of the wing, it was felt that a variable incidence angle covering the range of -15 to +90 degrees could be provided with only a minor weight penalty. This variable incidence was obtained by mounting the wing on a tubular spar, located at 35 percent wing chord to minimize pitching moments over most of the angle of attack range. Hydraulic servo actuation was, however, assumed. The variable incidence feature is desirable in that the vertical drag of the wing in hovering may be essentially eliminated. In forward flight, the variable incidence is equally desirable, at least for a research vehicle, so that the relative lifts of wing and rotor may be varied conveniently and thus optimized experimentally. The negative incidence feature may be desirable in moderate speed autorotation to avoid excessive positive angles of attack on the wing.

Because of the lower lift but higher propulsive force required of the rotor, in comparison to the basic HPH configuration, it is to be expected that a more forward rotor tilt is required. Forward tilt of the rotor resultant force is 7.7 degrees compared to 5.0 degrees for the basic HPH at a gross weight of 13,600 pounds at 175 knots. (Actual external drag values were used in this calculation rather than total equivalent drags - see Section III B). Tip path plane tilt is 8.8 degrees compared to 6.6 degrees for the basic HPH. Because of the 4 degree forward inclination of the main rotor shaft relative to the fuselage, forward flapping relative to the shaft is 4.8 degrees, well within design limits, if the fuselage is trimmed in a horizontal attitude. This flapping with respect to the shaft may be reduced if desired by trimming the fuselage (by means of the elevator) at a slight negative angle of attack. It is of interest to note that rotor blade flatwise vibratory stress may be determined from figure 35 for a lift of 11,000 pounds and an actual propulsive force required of 1490 pounds: a vibratory stress of approximately 3900 psi is obtained, which, as discussed in Section III D, should provide excellent blade life.

A comparison of the basic HPH and the HPH plus wing is shown in the following table.

Configuration	Basic HPH	HPH + 110 ft.^{2} wing (Modification (c))
Number of Blades	5	5
Design Cruise Speed, kt.	175	175
Tip Speed, ft./sec.	700	650
Parasite Area, ft.^{2}	13	13 + wing
Rotor Lift, lb.	13,600	11,000
Wing Lift, lb.	----	7000
Rotor System Horsepower	2230	2020
Weights, lb.		
Basic Weight Empty	8025	8025
Wing, Total	----	560
Trapped Fluids	115	115
Pilot(s)	200 (1)	400 (2)
Reserve Fuel	200	200
Operating Weight Empty	8540	9300
Fuel + Payload	5060	8700
Gross Weight	13,600	18,000
Payload for 100 nautical mile range, lb.	4300	7950

Note that weight provisions for two pilots were allowed for the HPH plus wing configuration. While there is every expectation that one pilot can learn to fly this configuration without aid, the variable incidence adds another independent control feature that would complicate flight procedures during a research investigation at least, and the second pilot was included because of the possibility that he might be considered desirable for all operations.

Payload-range and productivity characteristics of this modification are presented in figures 66 and 67 respectively. The HPH with wing carries a much higher payload at 175 knots than the basic HPH at the same speed and, for normal ranges, a slightly higher payload than the HPH cruising at 150 knots. As a result, the productivity of this modification is considerably higher than that of the basic HPH at either speed. The productivity, which exceeds 600 ton knots at 100 nautical miles range for a five minute turn-around time, is the highest of all configurations considered in this study. Because of the relative simplicity of this modification, it appears to offer the greatest potential gain in transport efficiency for a minimum of complication. The benefits of the wing are a result of the high cruising speed, which minimizes wing-rotor interference, and low parasite drag, which allows the rotor to propel the wing as well as the fuselage at relatively low tip speeds. The same advantages could not be expected from a wing on a less clean, slower helicopter.

It should be noted that the payload-range curve was calculated on the assumption that rotor lift is constant for the flight, with the wing unloaded as the aircraft weight decreases. It would not be possible to unload the rotor without a more forward rotor tilt required and, at some lower rotor lift limit, an inadequate propulsive force capability would result.

Despite the increased aerodynamic efficiency due to the addition of the wing, the ferry range of this modification is not significantly different from the basic HPH. At the relatively low speed at which maximum range is achieved, the wing is too small to offer a substantial increase in overall lift-drag ratio, and the slight gain in aerodynamic efficiency is approximately offset by the weight penalty of the wing.

D. MODIFICATION (d): HELICOPTER WITH WING AND TURBOJET ENGINES (JET COMPOUND)

As discussed previously, a helicopter rotor loses both lifting and propulsive force capacity as forward speed is increased. Consequently, the highest speed potential of an aircraft with a rotating wing is achieved by providing both a lifting wing and auxiliary propulsion. The resulting configuration is referred to herein as a compound helicopter. The fourth modification of the basic high performance helicopter consists of the addition of a wing and two turbojet engines, resulting in a compound helicopter configuration. Because jet engines rather than propellers are used, the configuration is referred to as a jet compound. The reasons for using the turbojets rather than some other propulsion scheme are the same as those discussed for modification (b). The jet compound version of the HPH, modification (d), is shown in figure 68.

Wing area selected for this modification was 200 square feet, nearly twice as large as the wing used in modification (c), partly because of a higher percentage of rotor unloading and partly because the jet compound is designed to fly at a considerable altitude as well as at sea level. It should be noted that modifications (a), (b), and (c) all depend on the rotor for lift and/or propulsion, so that altitude capability is relatively restricted by the rotor stall phenomenon. The compound helicopter, on the other hand, has no limitations with respect to altitude capability except those of a conventional fixed-wing airplane.

Two Pratt & Whitney JT-12 turbojets are used in this modification. With the main rotor in autorotation at a reduced rpm, the normal rated thrust of the two jets is sufficient to provide a cruising speed of 250 knots. Incidence of the jet engine nacelles is 5 degrees as opposed to 15 degrees on modification (b). Because the wing provides an efficient means of generating lift, the jet reaction lift is not important for this configuration, and the 5 degree incidence is provided primarily to reduce possible interference between the jet exhaust and tail surfaces.

The wing has an area of 200 square feet, a span of 35 feet, a taper ratio of 0.6, and an aspect ratio of 6.1. It was considered too large to tilt conveniently as in modification (c); consequently 40 percent full span flaps are indicated to reduce vertical drag in hovering. These flaps will also be useful in trimming fuselage angle of attack to desired values in high speed flight and, with upward deflection, avoiding wing stall in autorotational descents at moderate speeds. An estimate of the vertical drag of this wing in hovering with flaps deflected was made, including the increase in rotor thrust due to the presence of a horizontal surface as reported in Reference 6. The out-of-ground-effect net vertical drag of the wing was calculated to be 400 pounds at the normal rated power of the T-58-8 engines, reducing the maximum gross weight from 18,000 pounds to 17,600 pounds. The outboard portions of the wing flaps are used as ailerons in high speed forward flight, as the lateral control available from the rotor at reduced rpm may not be adequate.

Rotor tip speed is reduced to avoid the severe Mach number effects on rotor performance that would result at a flight speed of 250 knots at normal tip speed values. As reported in Reference 2, maximum aerodynamic efficiency would be obtained at rotor tip speeds on the order of 400 feet per second for a flight speed of 250 knots (advance ratio on the order of 1.0). To minimize the flapping sensitivity of the rotor (rate of change of flapping with angle of attack) and possible rpm control problems, it was felt that a somewhat higher tip speed would be desirable for an initial research investigation. A tip speed of 493 feet per second was selected for calculation of rotor characteristics. This tip speed provides an advance ratio (ratio of forward speed to tip speed) of 0.86 at 250 knots flight speed and an advancing tip Mach number of 0.82 at sea level. At the maximum altitude considered, 25,000 feet, the speed of sound is reduced to 91 percent of the standard sea level value, and the above combination of forward and tip speeds results in an advancing tip Mach number of 0.90, the maximum value considered acceptable for this rotor.

The performance study indicated that autorotation of the rotor at a lift of 6000 pounds at 250 knots at sea level is near the optimum operating condition. Collective pitch ($\theta_{.75R}$) and control axis angle of attack are -1.0 and +3.0 degrees respectively. Autorotation is convenient in the sense that no gear ratio change between engine and rotor is required at the reduced rotor rpm. The S-61 transmission utilized incorporates a free wheeling unit that uncouples the engines completely from the rotor when the engines are stopped. On the other hand, gearing the rotor to the engine and powering the rotor to some extent might be desirable from the standpoint of controlling rpm or controlling tip path plane angle of attack and flapping relative to the shaft. A much more detailed study of rotor operation and control is required for the compound helicopter; the present study is intended only to evaluate some of the performance potentialities of the configuration.

The drag of the rotor under the above conditions of autorotation at 6000 pounds of lift is 650 pounds, resulting in a rotor lift-drag ratio of 9.2. In terms of parasite area, the rotor drag is equivalent to 3.2 square feet of parasite area, a relatively small penalty considering that the rotor produces useful lift. At altitude, it is assumed that the rotor angle of attack and pitch settings are unchanged, so that the rotor is unloaded as atmospheric density decreases.

A comparison of the jet compound with the basic high performance helicopter is shown in the table below.

Configuration	Basic HPH	Jet Compound HPH + 200 ft.^{2} wing + 2 JT-12 Turbojets (Modification (d))	
Number of Blades	5	5	
Design Cruise Speed, kt.	175	250	
Tip Speed, ft./sec.	700	493	
Parasite Area, ft.^{2}	13	15 (+ wing & rotor blades)	
Altitude, ft.	0	0	25,000
Rotor Horsepower	2230	0	0
Rotor Lift, lb.	13,600	6000	2700
Wing Lift, lb.	----	11,600	14,900
Wing Lift Coefficient	----	0.27	0.78
Jet Thrust, lb.	----	4350	2240
Weights, lb.			
Basic Weight Empty	8025	8025	8025
Jet Engines (Installed)	----	1800	1800
Wing	----	800	800
Pilot(s)	200 (1)	400 (2)	400 (2)
Trapped Fluids	115	115	115
Reserve Fuel	200	200	200
Oxygen Equipment	----	----	50
Operating Weight Empty	8540	11,340	11,390
Fuel + Payload	5060	6260	6210
Gross Weight	13,600	17,600	17,600
Payload for 100 nautical mile range	4300	4350	5200

The payload-range curves for this modification are shown in figure 69. Payload of the jet compound at sea level becomes inferior to that of the basic HPH cruising at 175 knots at ranges greater than 100 nautical miles due to the high fuel consumption of the turbojets. At 25,000 feet altitude, payload is superior out to about 500 nautical miles. Payload of the jet compound at any range or altitude is considerably inferior to that of the basic HPH cruising at 150 knots. It should be noted, however, that this result is due to the hovering power limitation; if the thrust power available from the turbojets in forward flight were made available to the rotor system in hovering, payloads could be substantially greater, since the gross weight of the compound is not limited by the high speed flight condition.

Productivity of the jet compound is shown in figure 70. Because of the high speed of this configuration, the productivity at short ranges, even at sea level, is superior to the basic HPH at 175 knots, and is nearly equal to the HPH cruising at 150 knots. At an altitude of 25,000 feet, the productivity of the jet compound exceeds that of the HPH at 150 knots for ranges up to nearly 200 nautical miles. The rapid drop of productivity with range for the jet compound at sea level, of course, is due to the high fuel consumption of the turbojets. If the auxiliary powerplants were turboshaft engines driving propellers, and if these turboshaft engines were also geared to the main rotor in hovering and slow speed flight, then payload capacity might be significantly higher and range capability greatly extended. Productivity of the compound helicopter then might very well be greater than for any other configuration considered, at the expense of a heavier, more complicated aircraft. The present jet compound is intended primarily as a relatively inexpensive research version of the compound helicopter concept. It is significant, however, that very useful payloads and productivity figures are generated with this aircraft, at a speed very much higher than for any other current rotary wing VTOL.

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2. Fradenburgh, Evan A., Aerodynamic Efficiency Potentials of Rotary Wing Aircraft, paper presented before the A. H. S. 16th Annual National Forum, Washington, D. C., May 11-14, 1960.

3. Michel, Philip L., Research and Design Progress Toward High Performance Rotary Wing Aircraft, paper presented before the I. A. S. 28th Annual Meeting, New York, New York, January 25-27, 1960.

4. U. S. Army Transportation Research Command Contract No. DA 44-177-TC-548, Forward Flight Tests of High Speed Rotor Blades, contract date June 24, 1959.

5. Gessow, Alfred, and Gustafson, F. B., Effect of Blade Cutout on Power Required by Helicopters Operating at High Tip-Speed Ratios, NASA Technical Note D-382, September, 1960.

6. Fradenburgh, Evan A., The Helicopter and the Ground Effect Machine, A. H. S. Journal, October, 1960.

7. Amer, Kennth B., Theory of Helicopter Damping in Pitch or Roll and a Comparison with Flight Measurements, NACA TN 2136, October, 1950.

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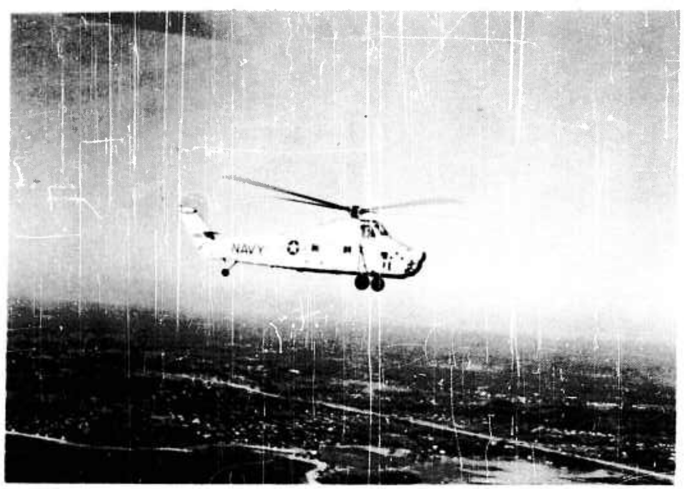
S-59 (XH-39)

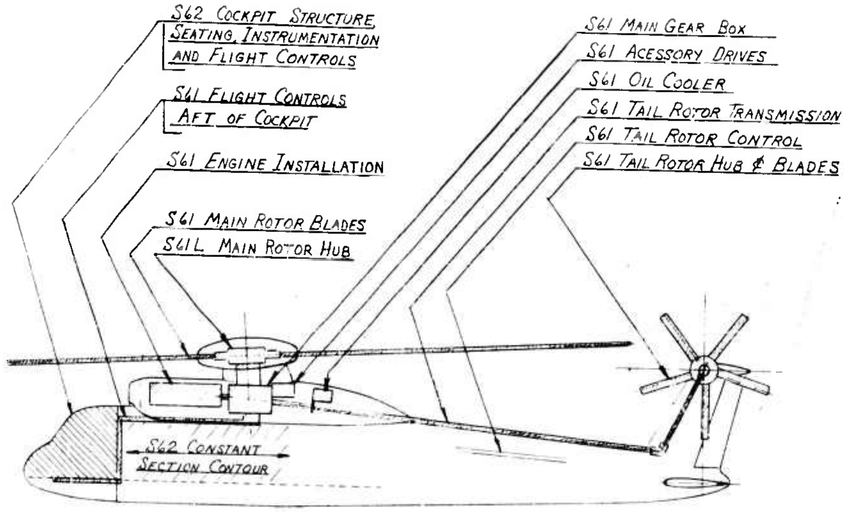
S-62



HSS-1F

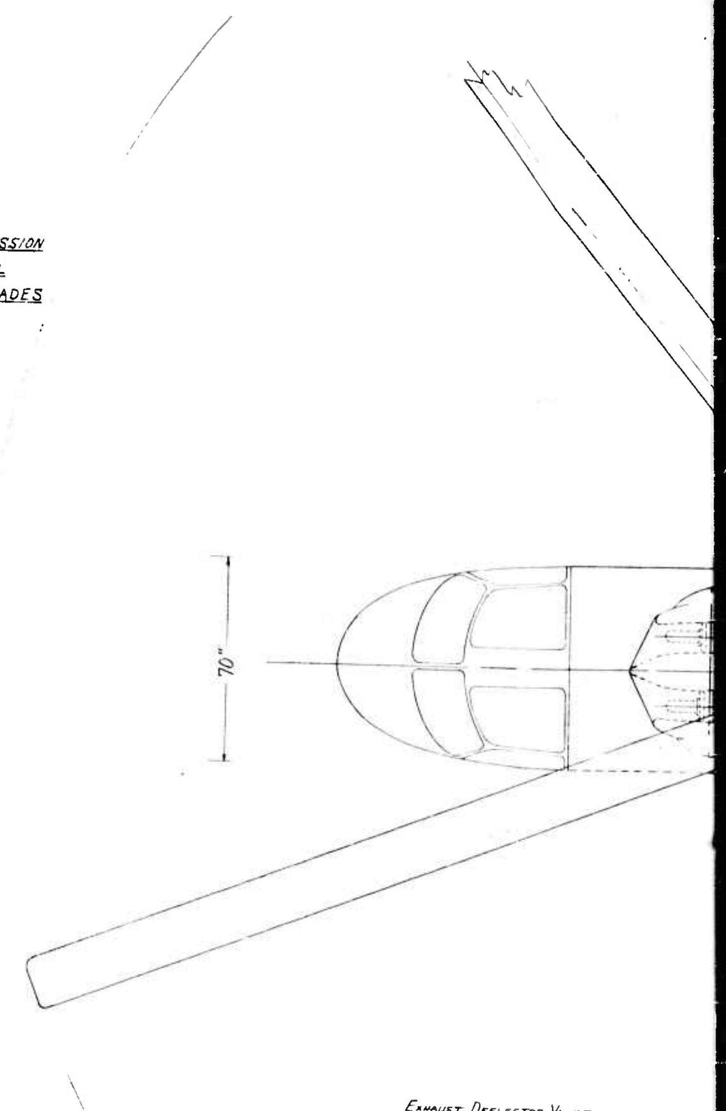
S-61 (HSS-2)

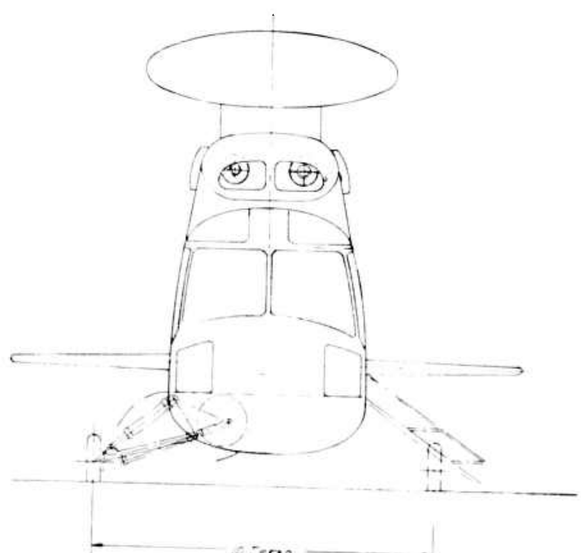
FIGURE 7. AIRCRAFT CONSIDERED IN DESIGN STUDY

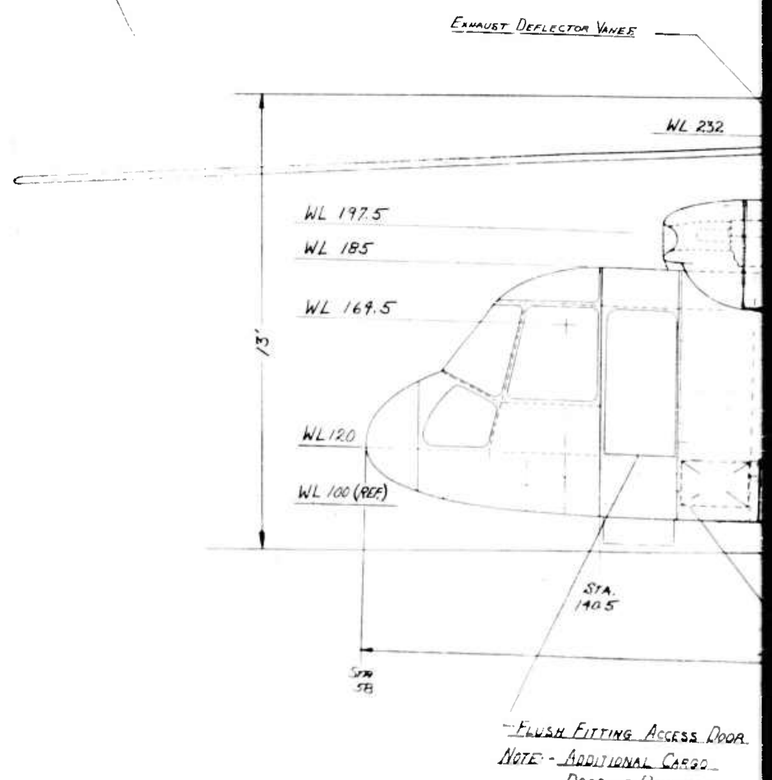


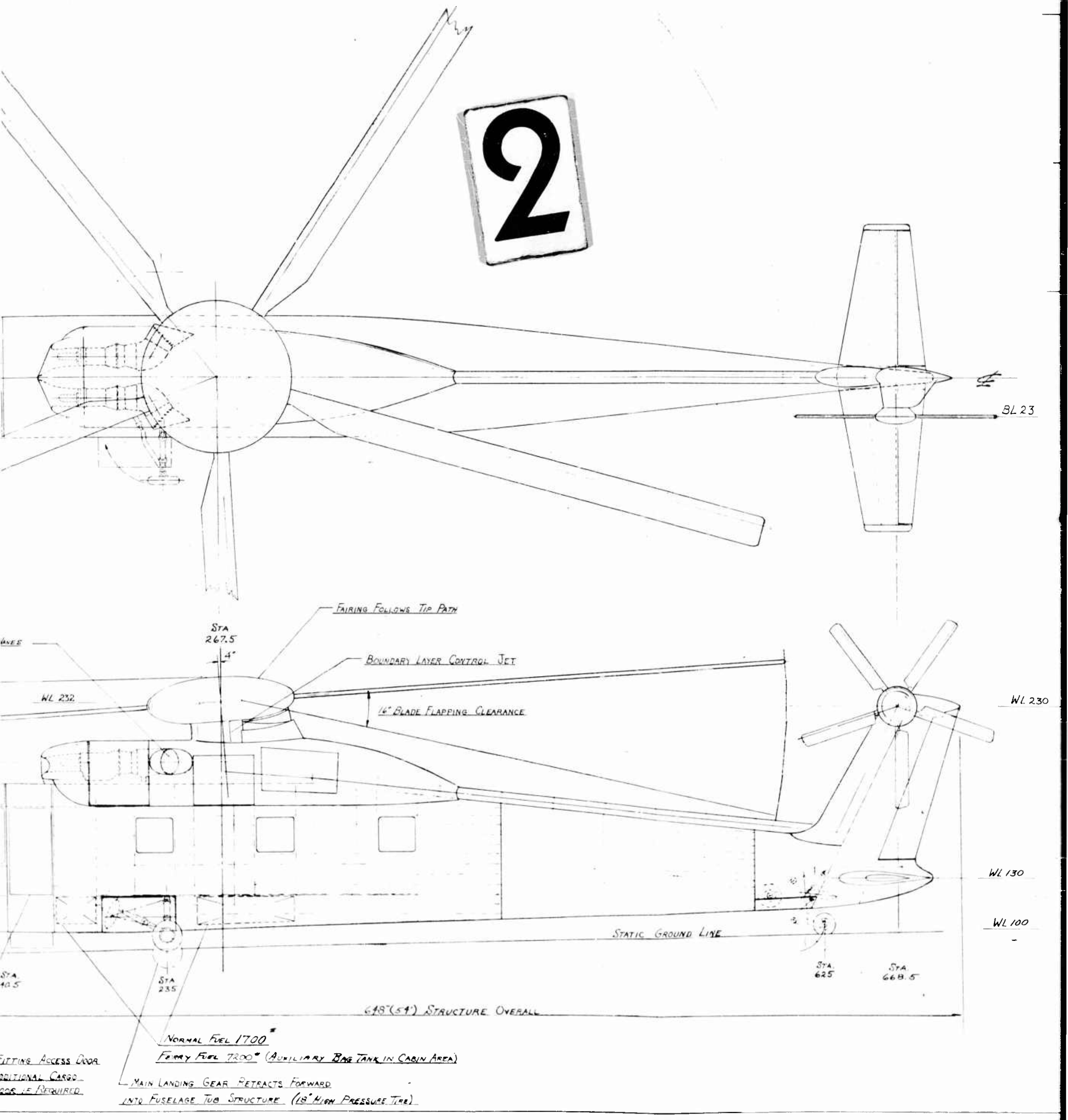
AVAILIBILITY OF EXISTING COMPONENTS

1

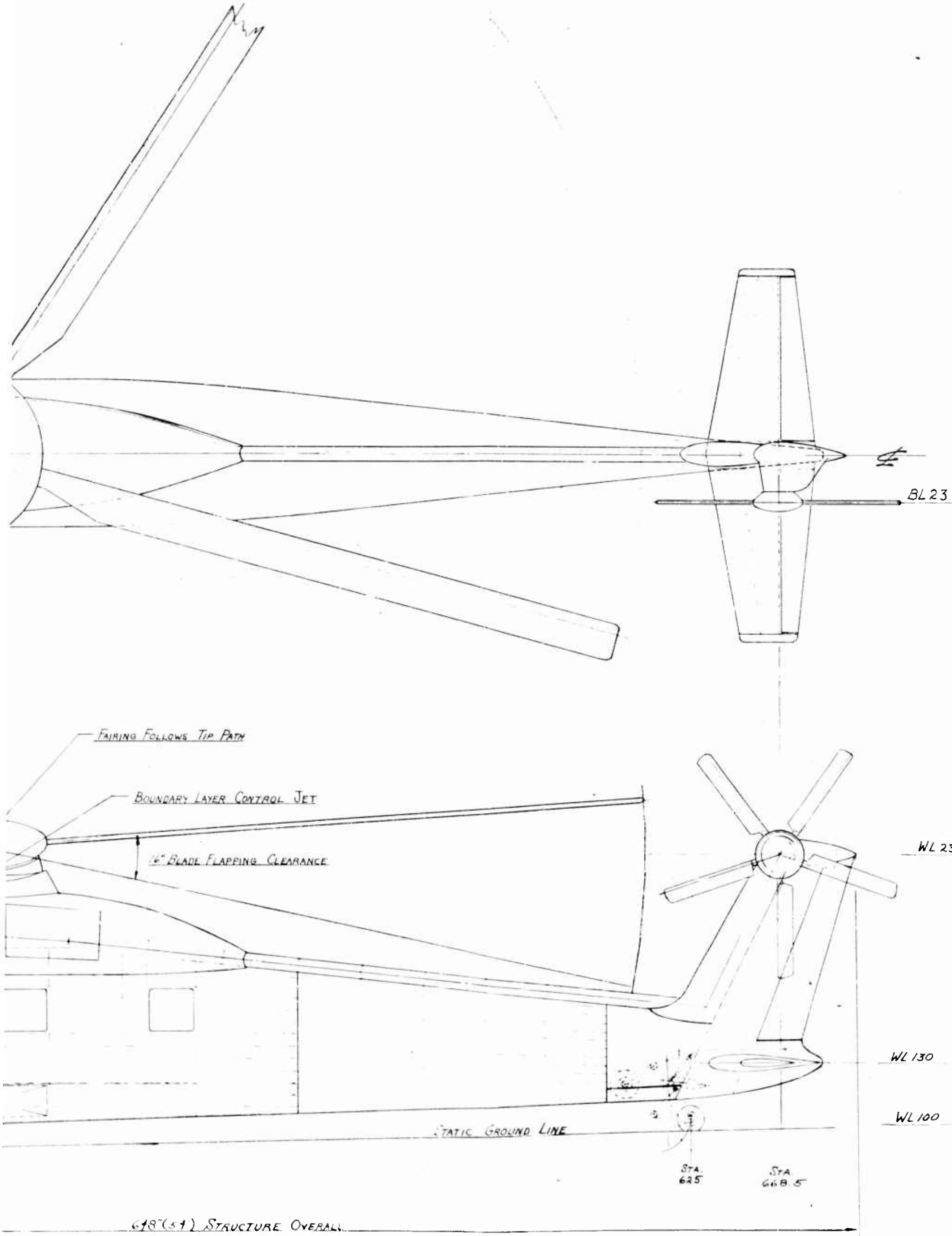








2
BL 23
FAIRING FOLLOWS TIP PATH
STA 267.5
4°
BOUNDARY LAYER CONTROL JET
WL 232
16" BLADE FLAPPING CLEARANCE
WL 230
WL 130
WL 100
STATIC GROUND LINE
STA 235
STA 625
STA 668.5
648"(54') STRUCTURE OVERALL
NORMAL FUEL 1700#
FERRY FUEL 7200# (AUXILIARY BAG TANK IN CABIN AREA)
MAIN LANDING GEAR RETRACTS FORWARD
INTO FUSELAGE TUB STRUCTURE (18" HIGH PRESSURE TIRE)



Main Rotor

Diameter	56 ft
No. Blades	5
Chord	18.25 in
Solidity	.086
Twist	-4°

Tail Rotor

Diameter	10 ft
No. Blades	5
Chord	7.35 in
Solidity	.195
Twist	0°

Horizontal Tail

Area - Total	50 ft^2
- Elevators	15 ft^2
Section	NACA 0015
Incidence	0°

Fin and Rudder

Area - Total	40 ft^2
- Rudder	10 ft^2
Section	NACA 0020

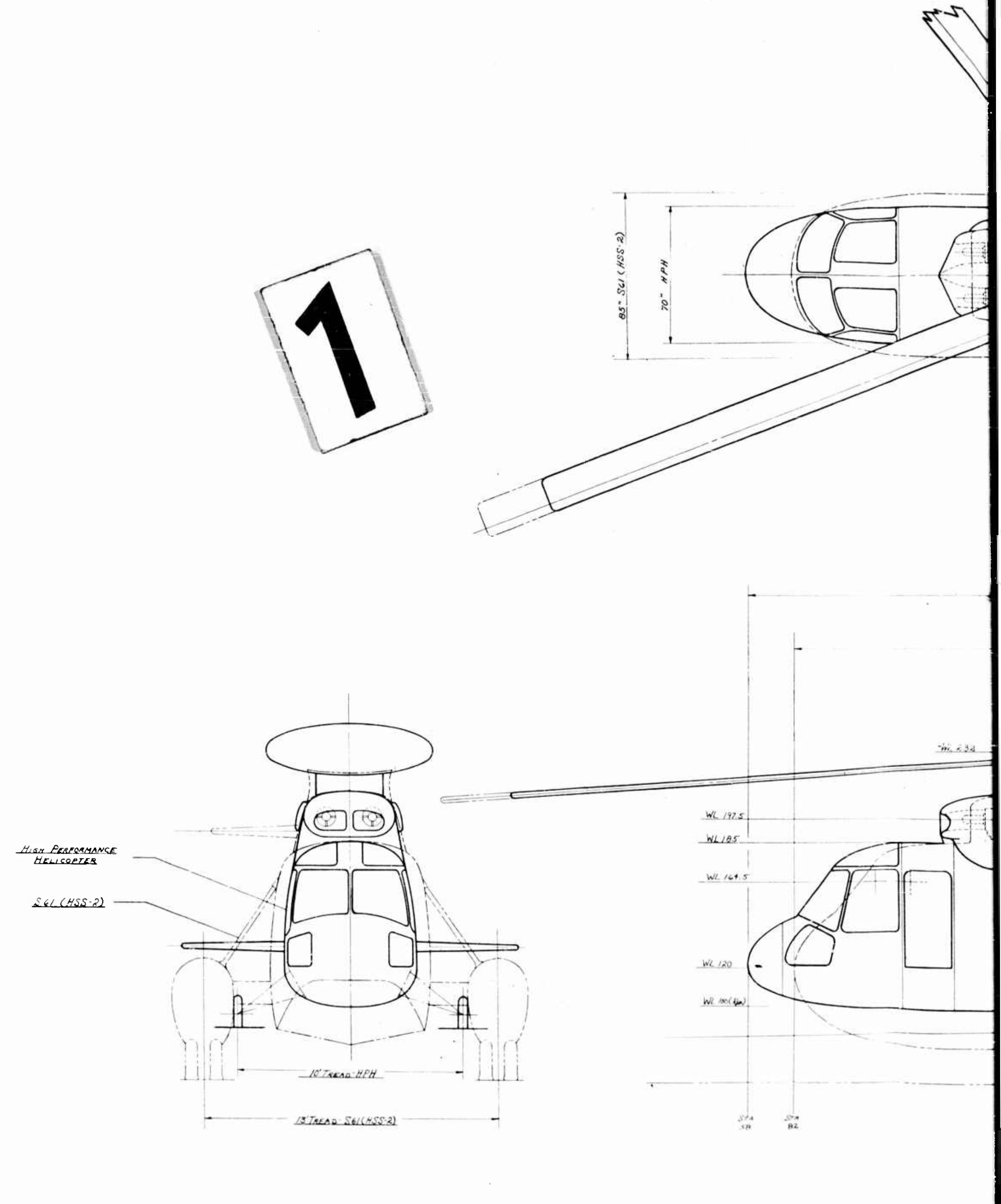
Power Unit

2 GE - T 58 - 8 Engines

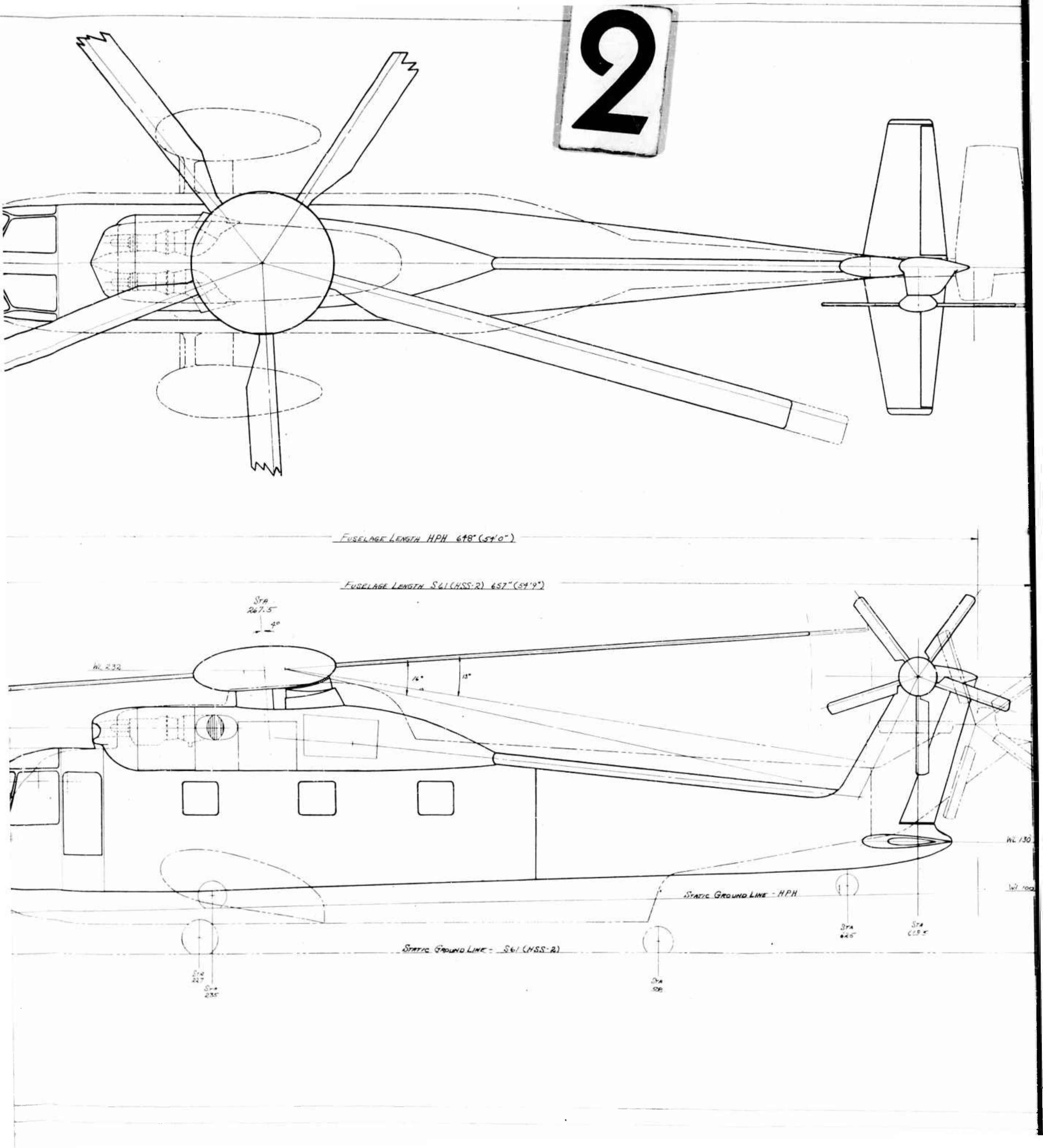
3

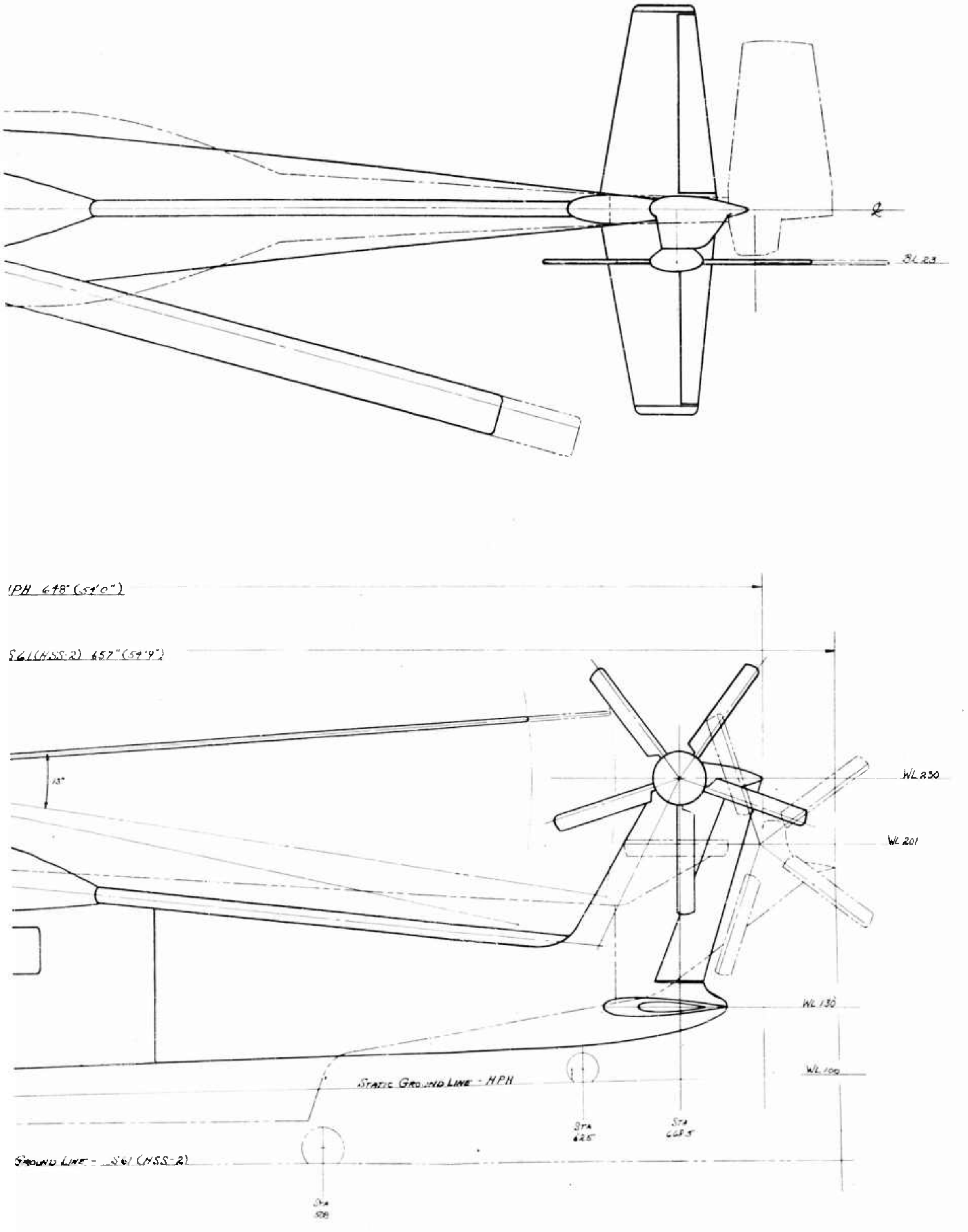
2*
" (Auxiliary Bag Tank in Cabin Area)
racts Forward
ture (18" High Pressure Tire)

Figure 8 General Arrangement High Performance Helicopter	Sikorsky Aircraft Stratford, Conn. Division of United Aircraft Corporation Page 73



1
85" S61 (HSS-2)
70" HPH
HIGH PERFORMANCE HELICOPTER
S61 (HSS-2)
10' TREAD HPH
13' TREAD S61(HSS-2)
WL 197.5
WL 185
WL 164.5
WL 120
STA 58
STA 82

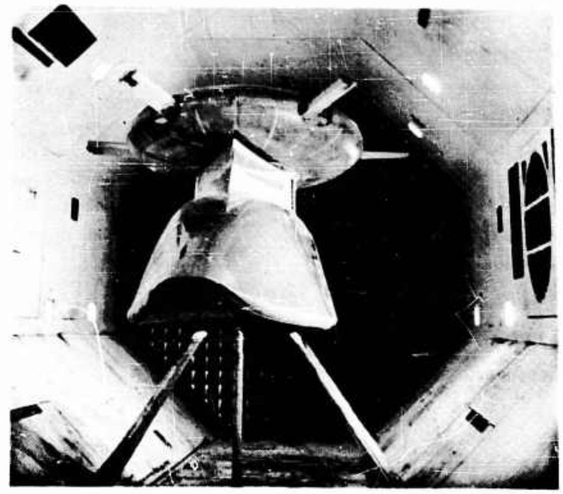

2
FUSELAGE LENGTH HPH 648" (54'0")
FUSELAGE LENGTH S61 (HSS-2) 657" (54'9")
STA 267.5
4°
WL 232
16°
13°
WL 130
WL 100
STATIC GROUND LINE - HPH
STA 625
STATIC GROUND LINE - S61 (HSS-2)
STA 227
STA 235



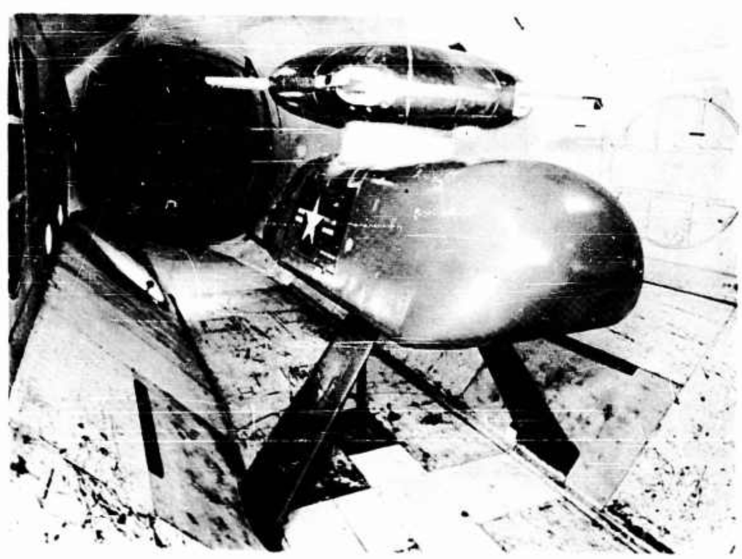
3

FIGURE 9
DEMENSIONAL COMPARISON
HIGH PERFORMANCE
HELICOPTER WITH S61(HSS-2)

SIKORSKY AIRCRAFT
STRATFORD, CONN.
DIVISION OF
UNITED AIRCRAFT CORPORATION



Rear View of Model



Front View of Model

FIGURE 10. FULL SCALE ROTOR HEAD FAIRING-PYLON MODEL

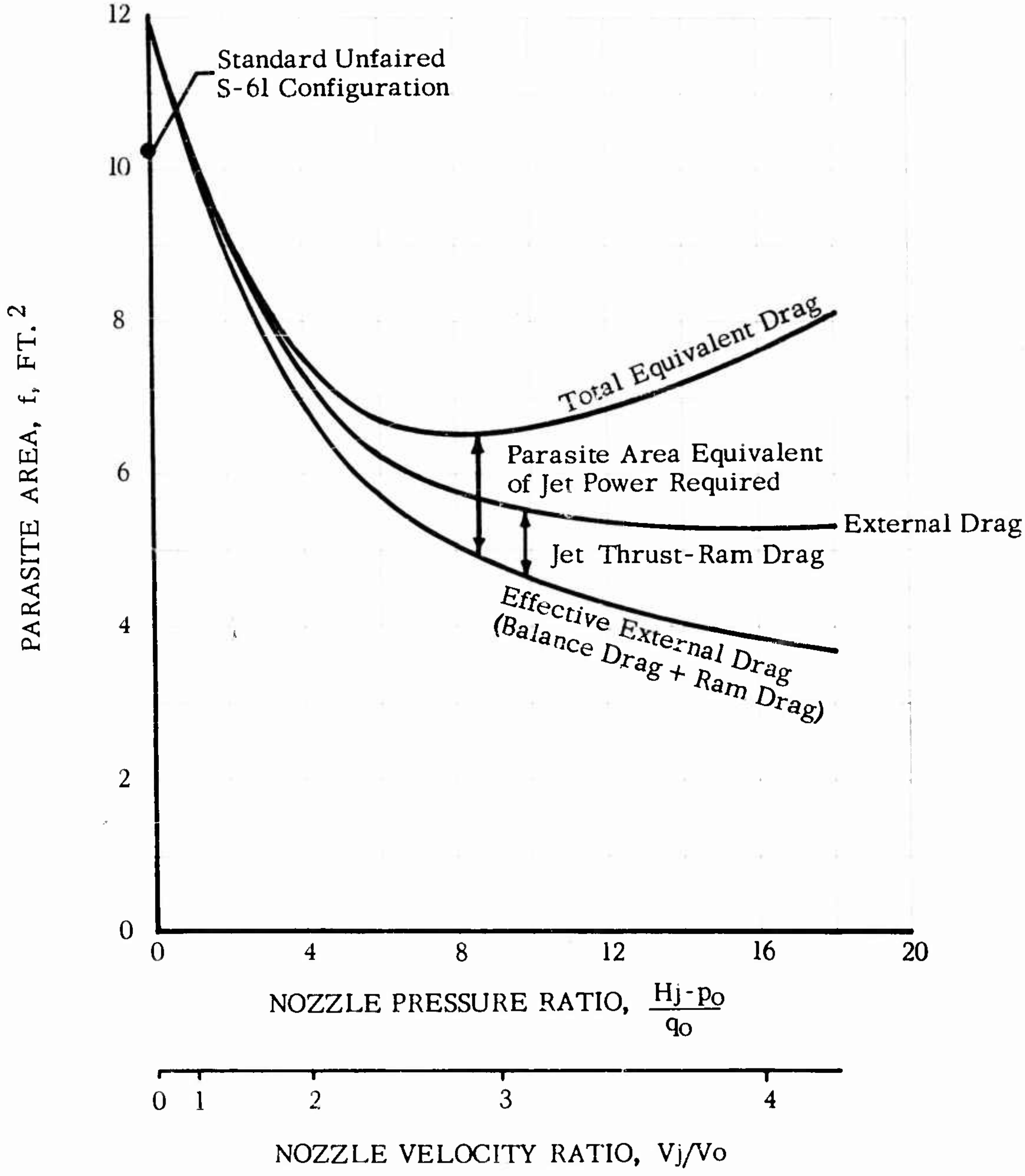


FIGURE 11. RESULTS OF ROTOR HEAD FAIRING-PYLON MODEL TEST

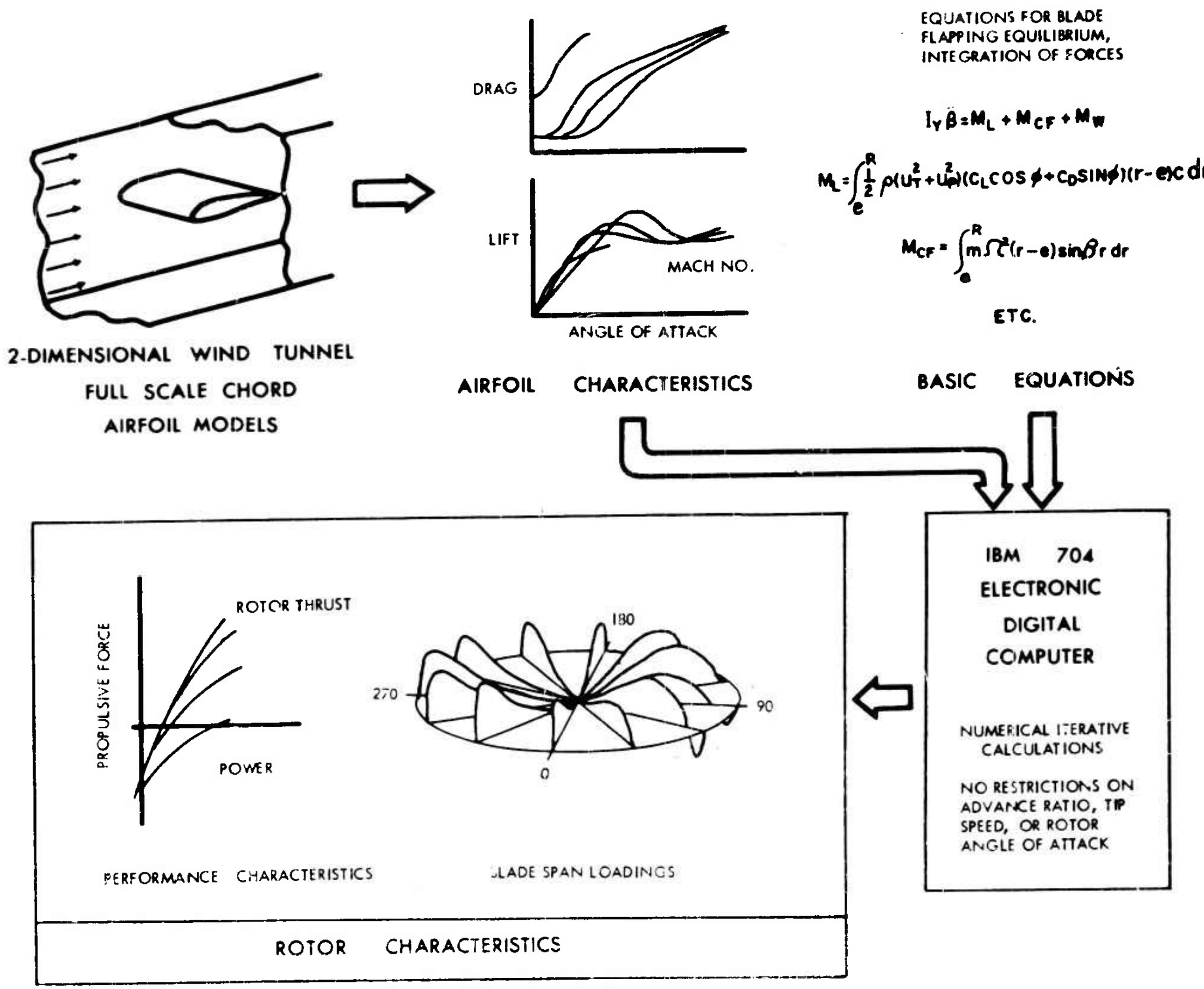


FIGURE 12. NUMERICAL INTEGRATION ROTOR PERFORMANCE METHOD

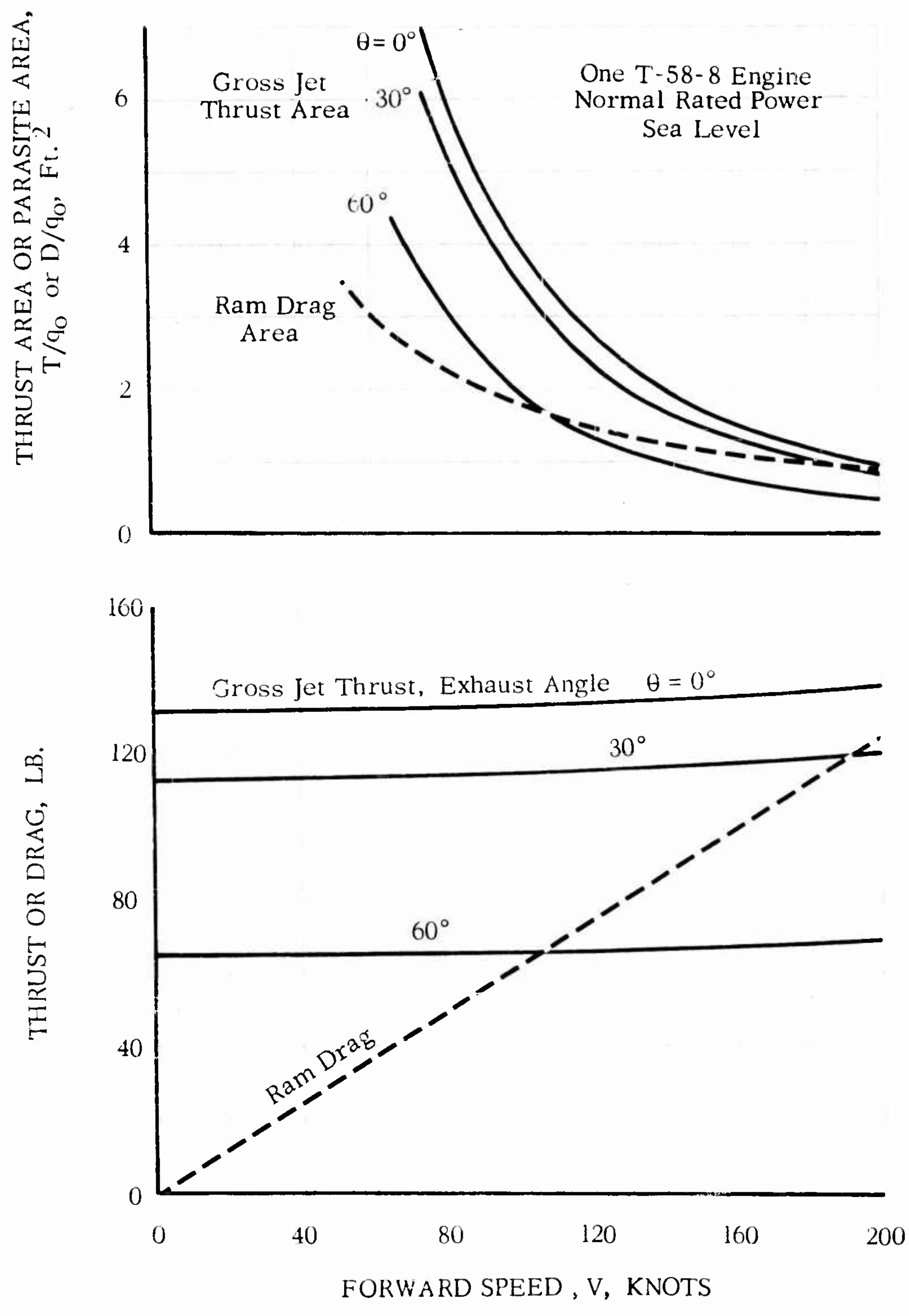


FIGURE 13. ENGINE AIRFLOW EXTERNAL FORCES

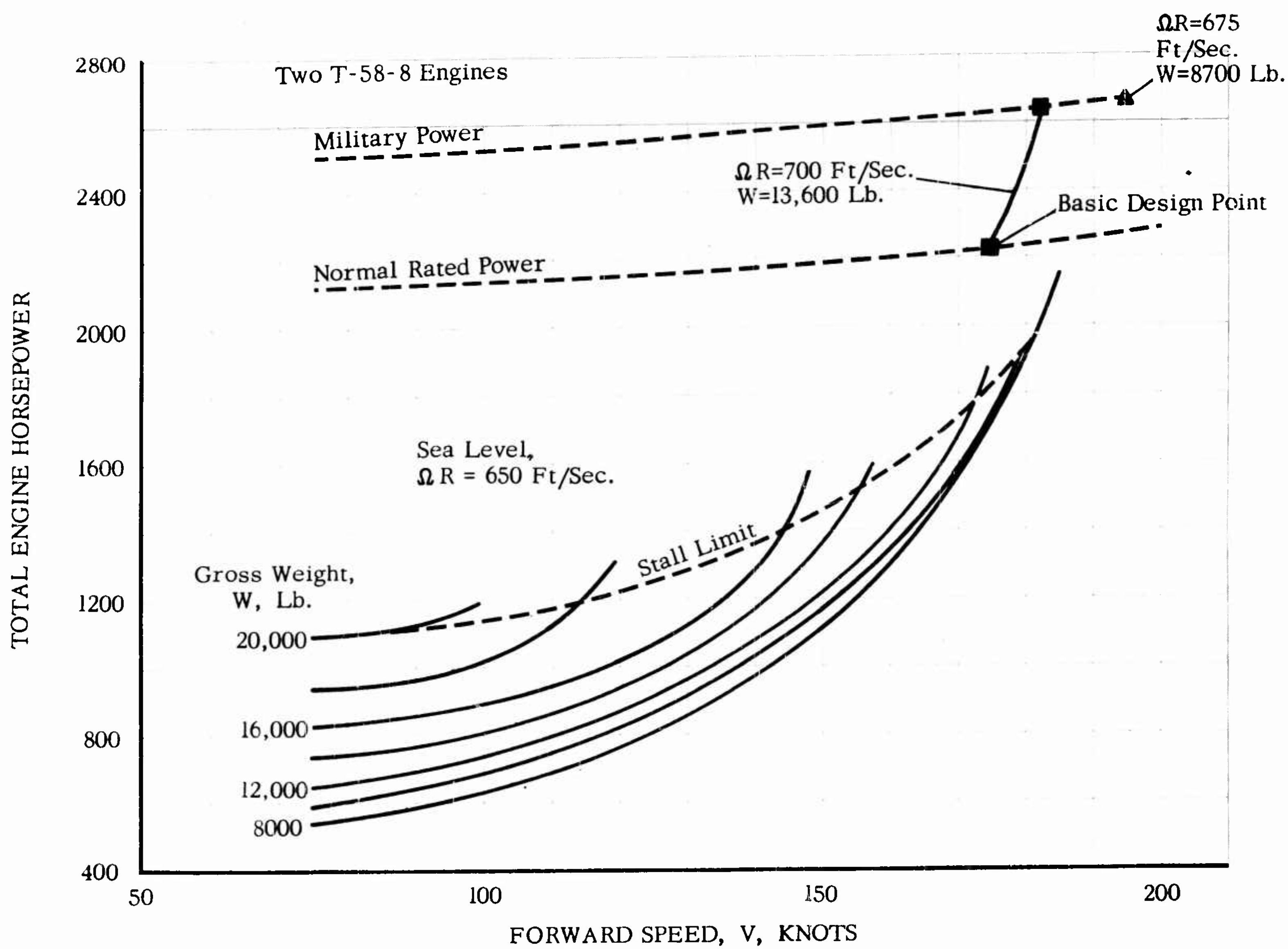


FIGURE 14. CALCULATED FORWARD FLIGHT PERFORMANCE-SEA LEVEL

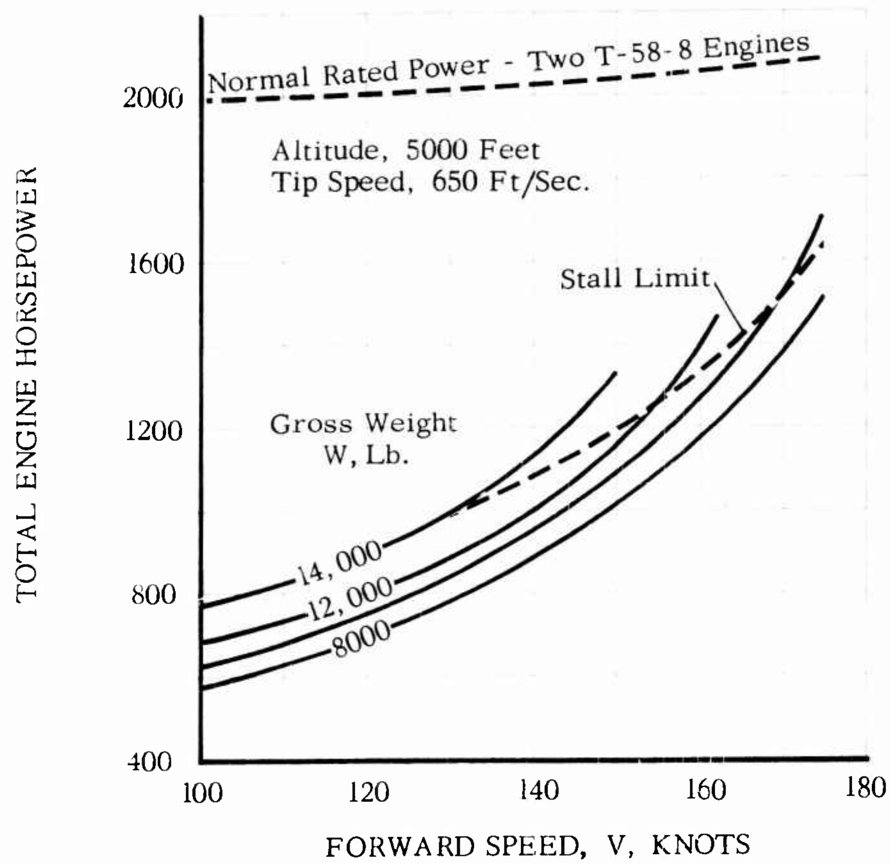


FIGURE 15. CALCULATED FORWARD FLIGHT PERFORMANCE-5000 FEET

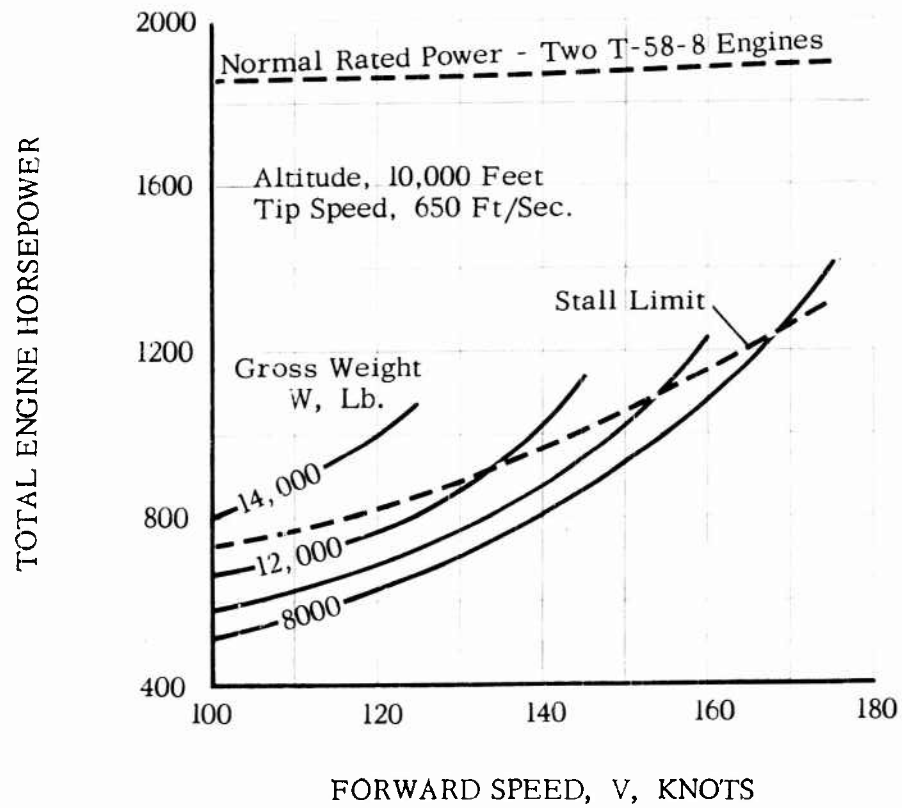


FIGURE 16. CALCULATED FORWARD FLIGHT PERFORMANCE-10,000 FEET

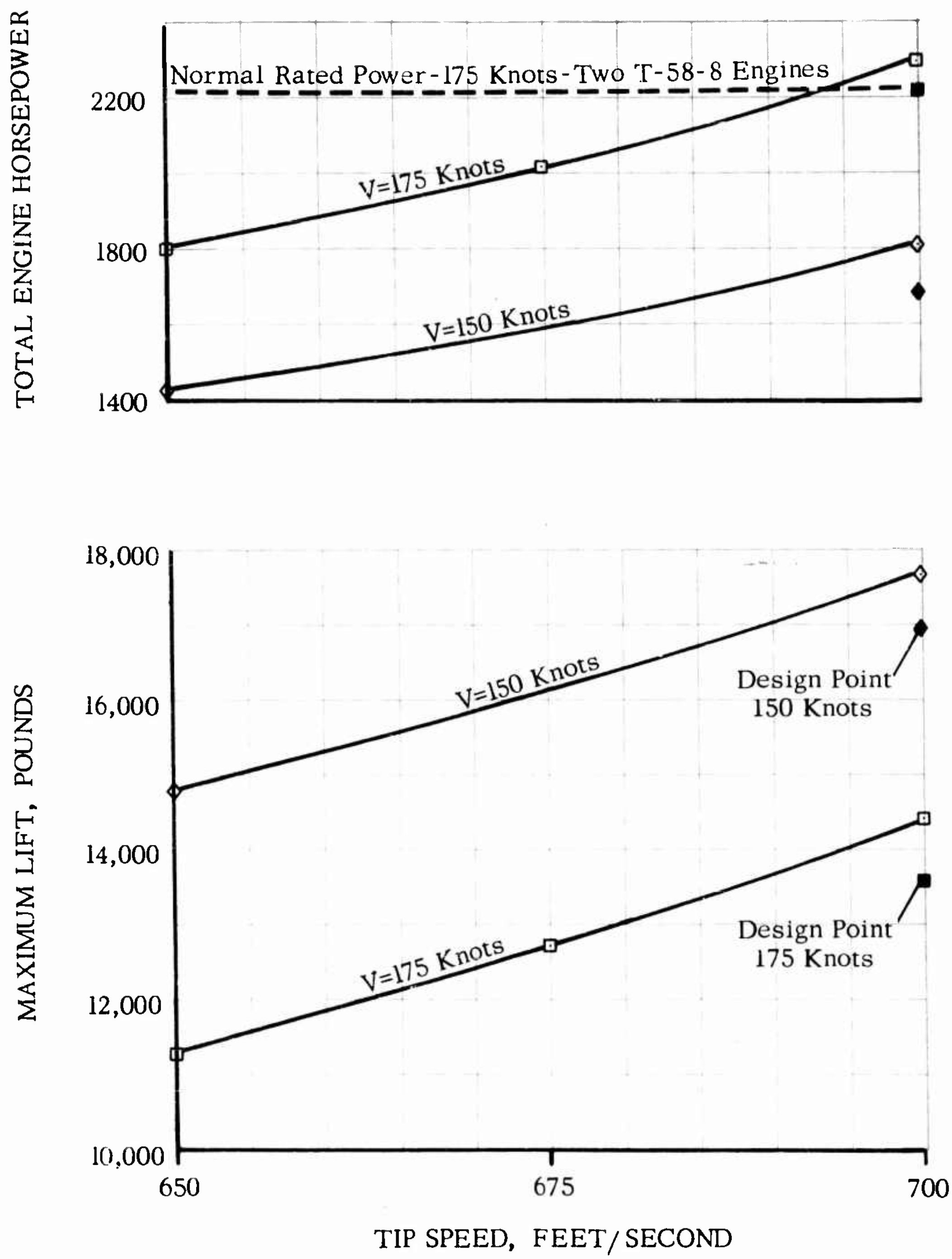


FIGURE 17. EFFECT OF TIP SPEED ON MAXIMUM LIFT CAPABILITY

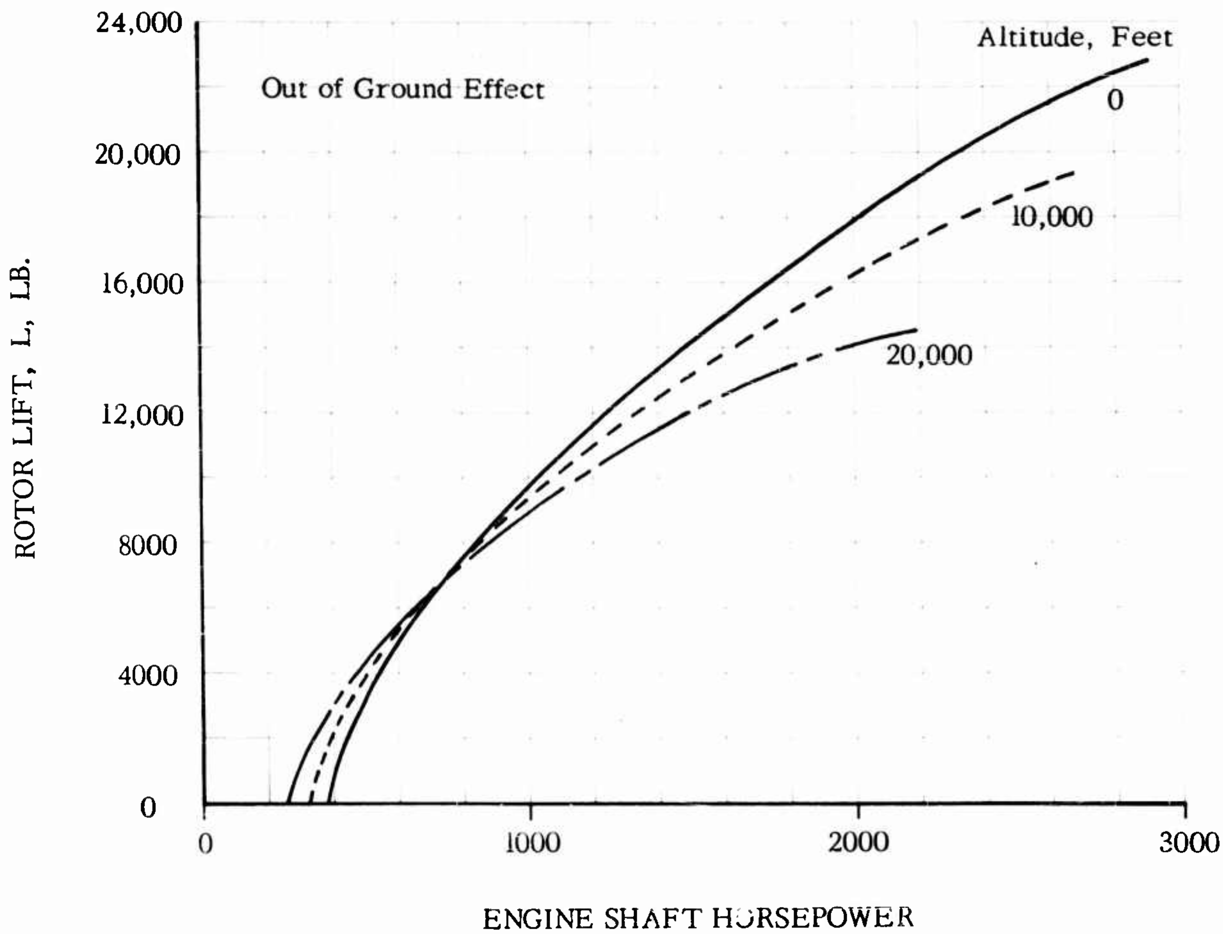


FIGURE 18. CALCULATED HOVERING PERFORMANCE

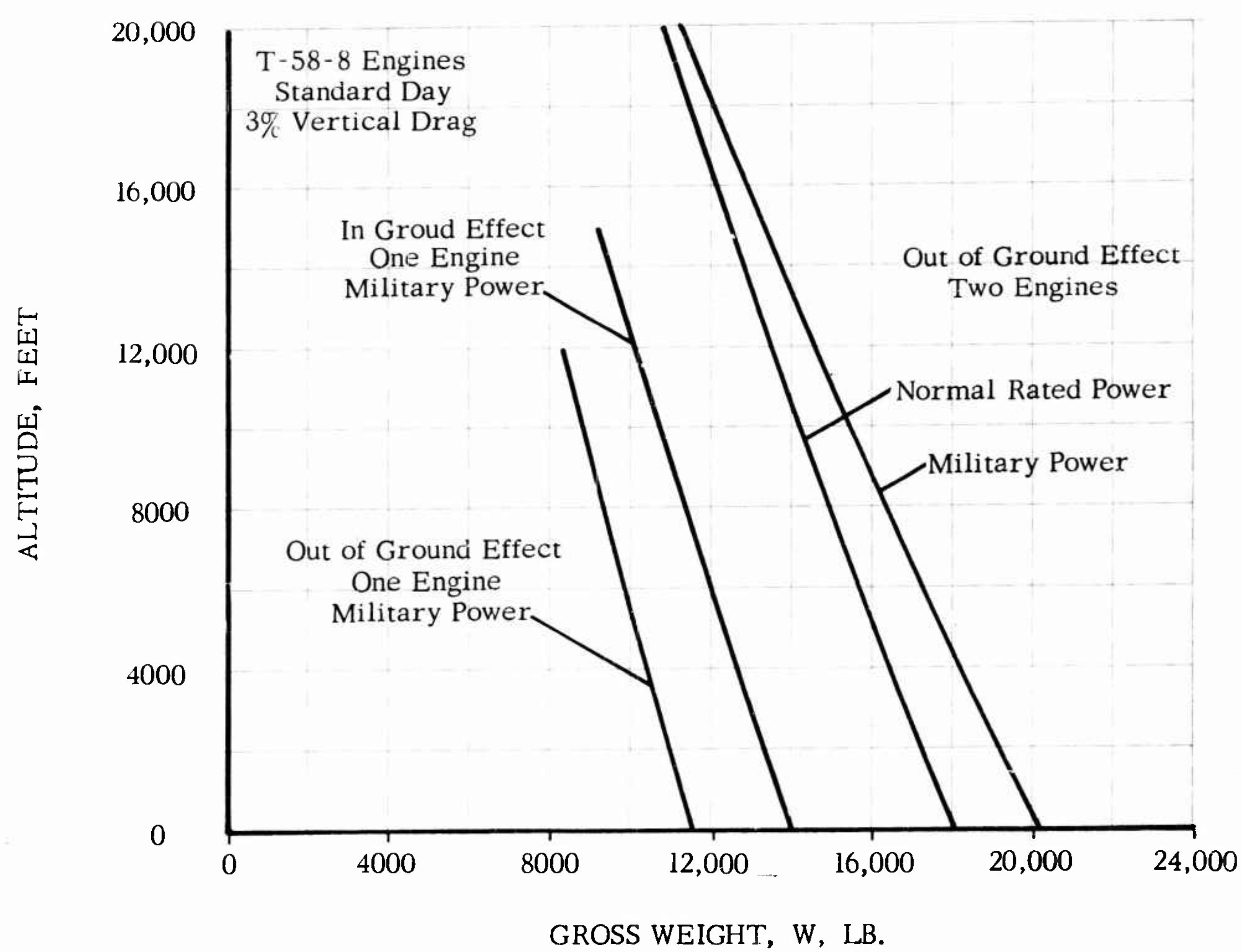


FIGURE 19. HOVERING CEILINGS

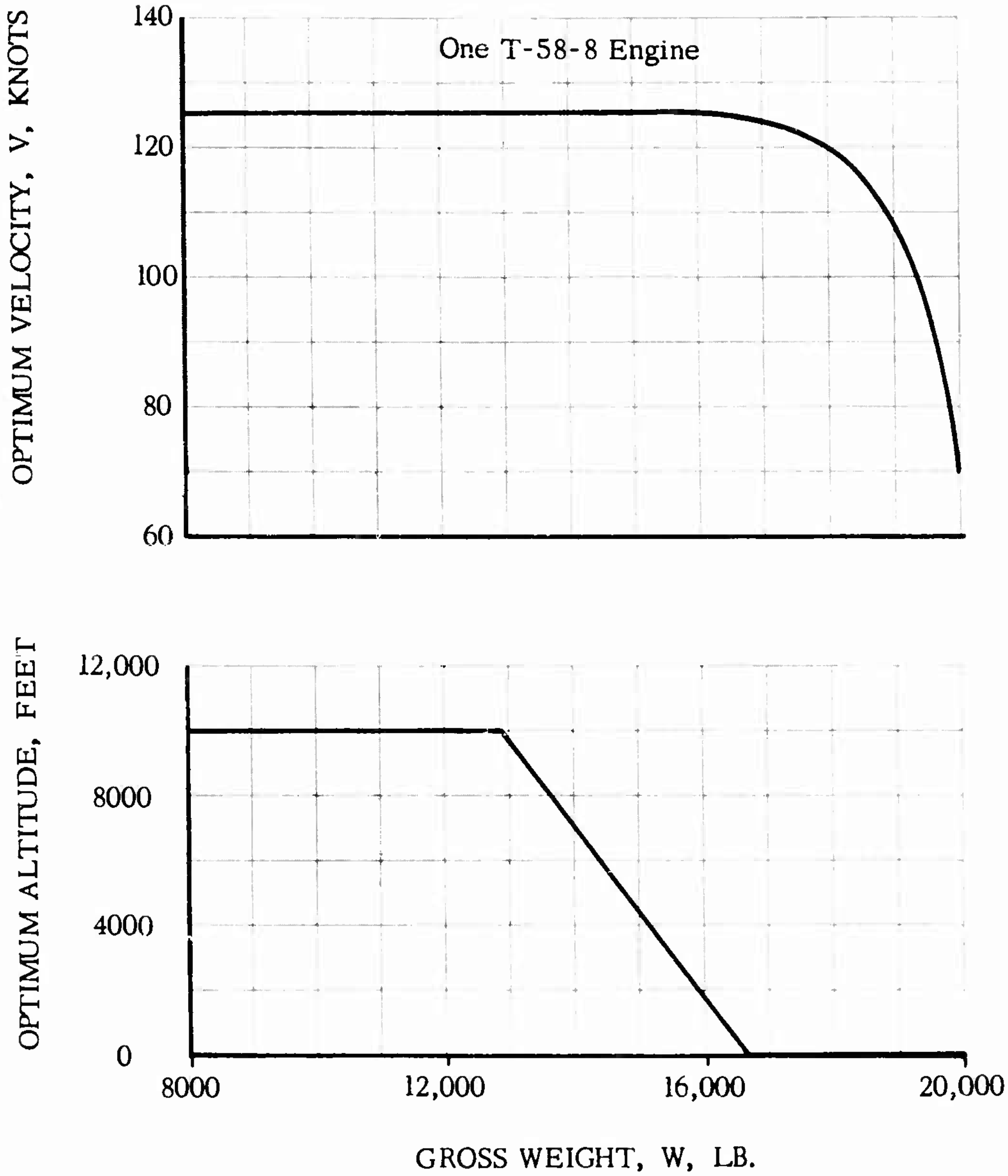


FIGURE 20. FLIGHT CONDITIONS FOR MAXIMUM RANGE

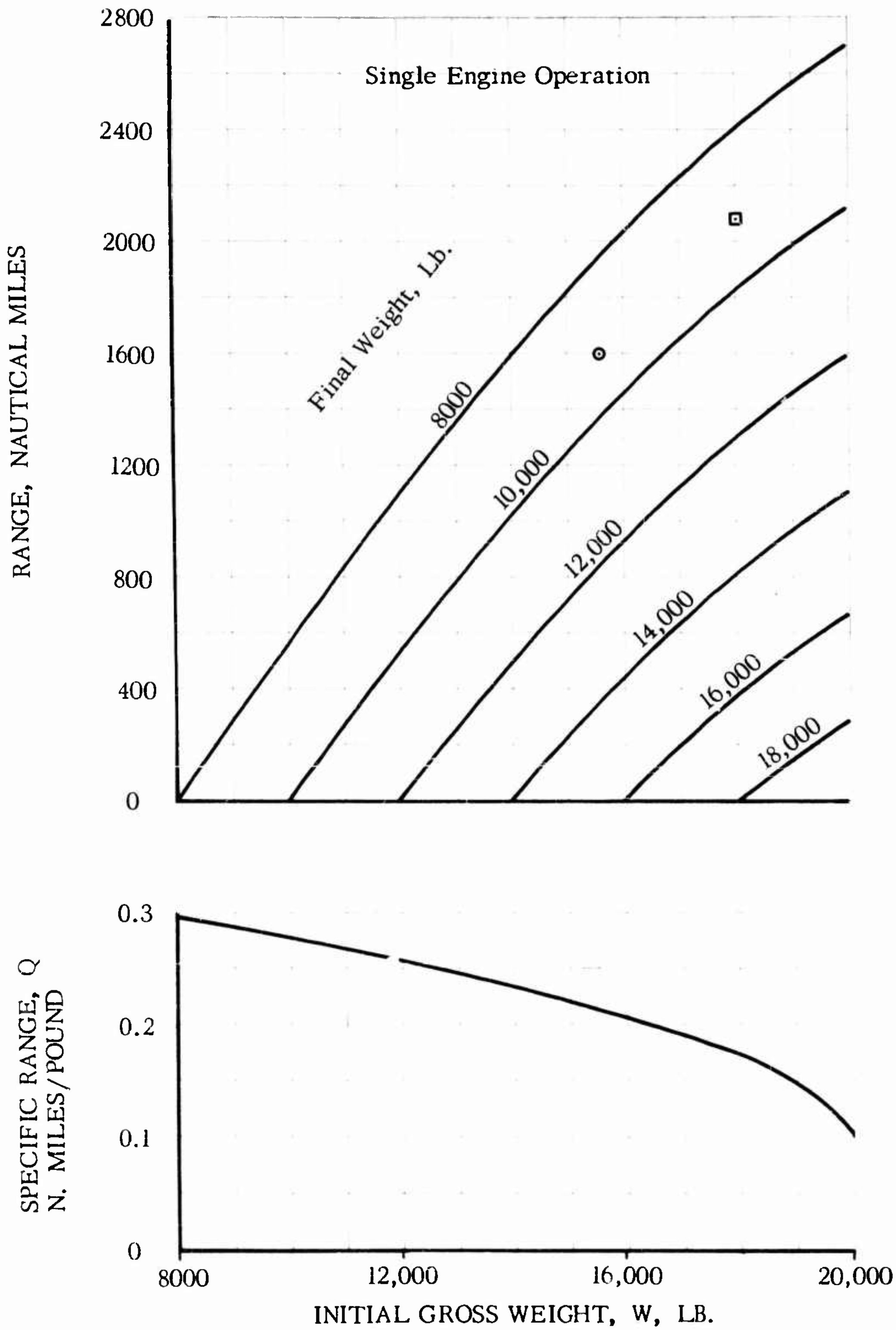


FIGURE 21. FERRY RANGE CHARACTERISTICS OF HPH

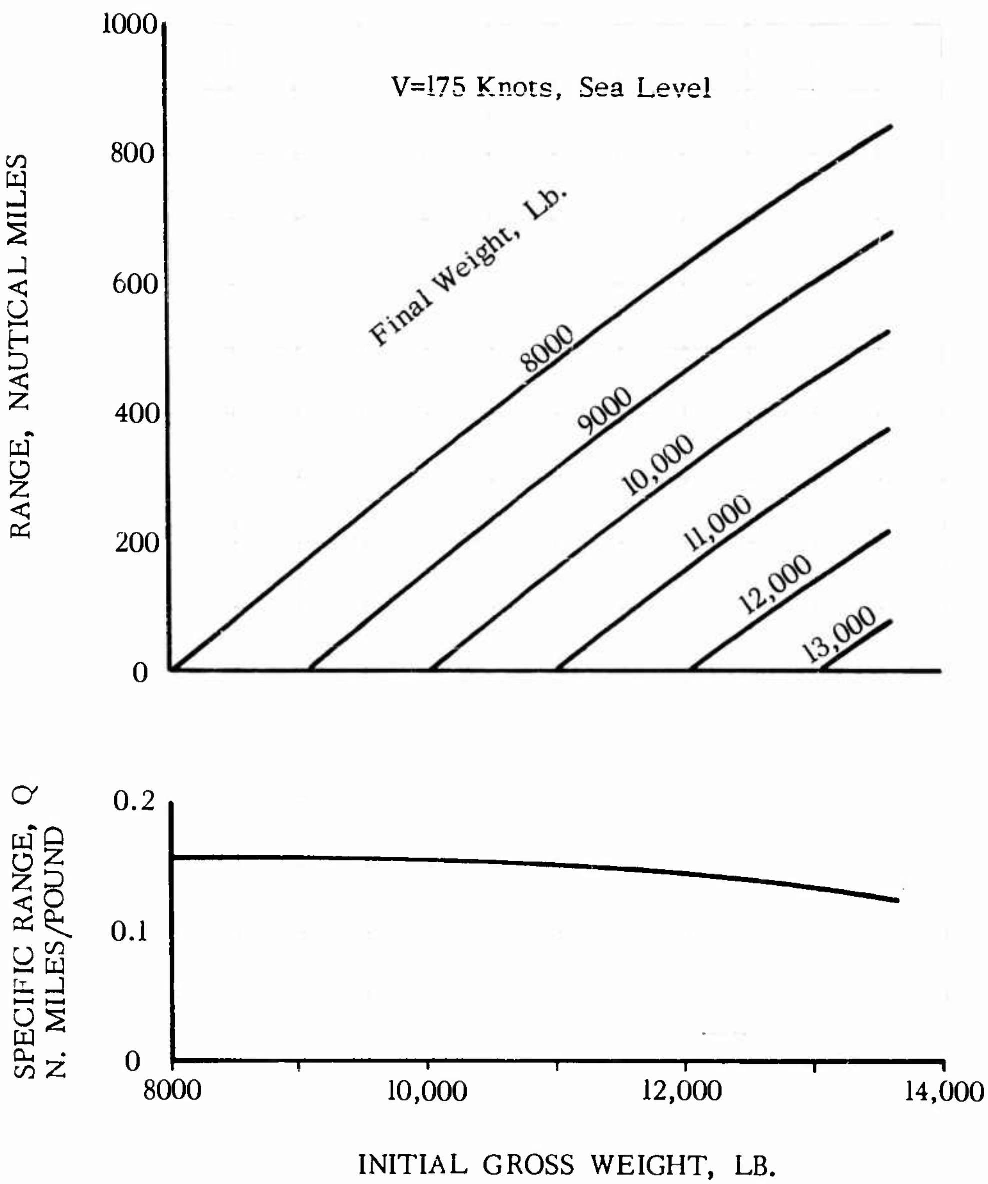


FIGURE 22. RANGE CHARACTERISTICS AT 175 KNOTS

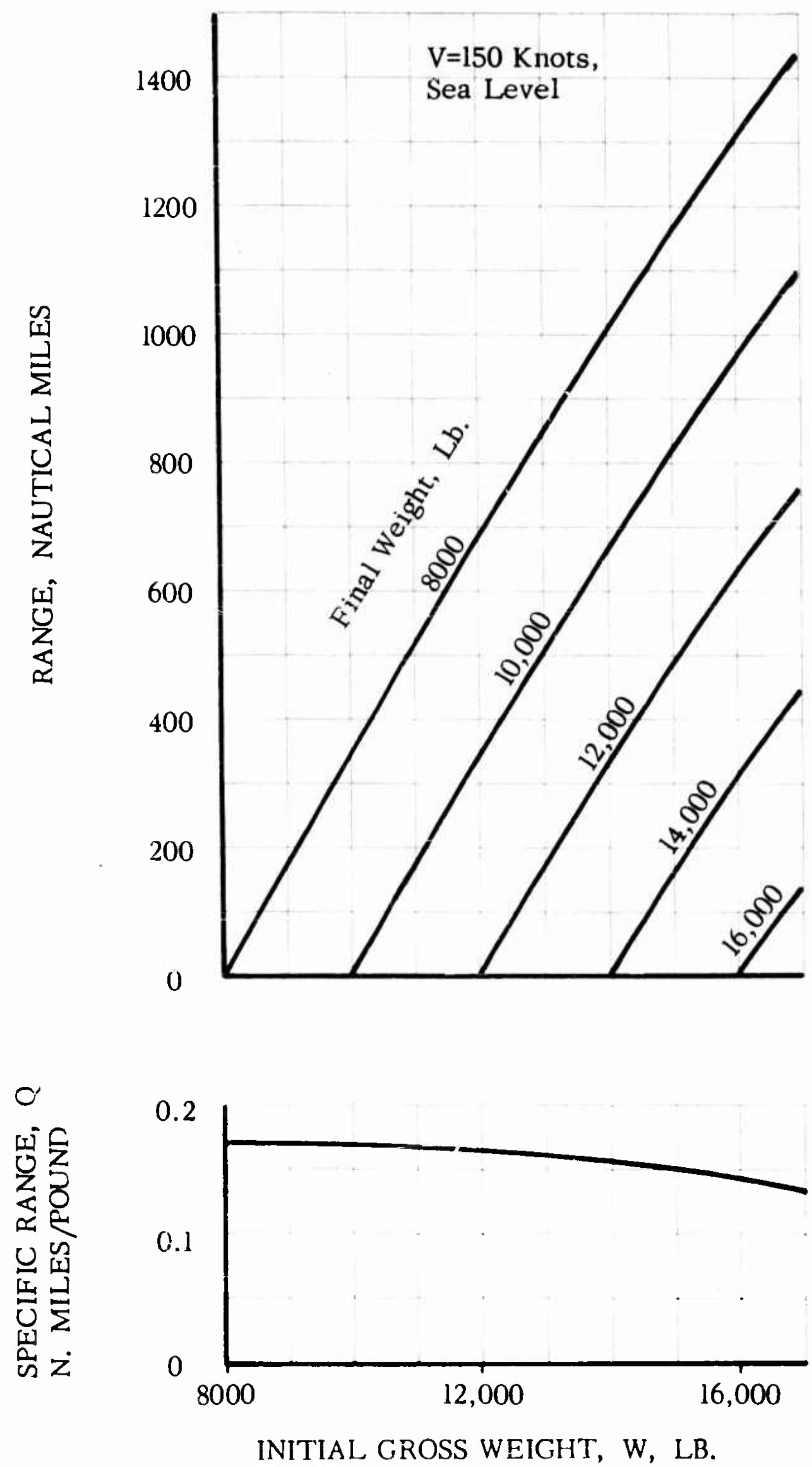
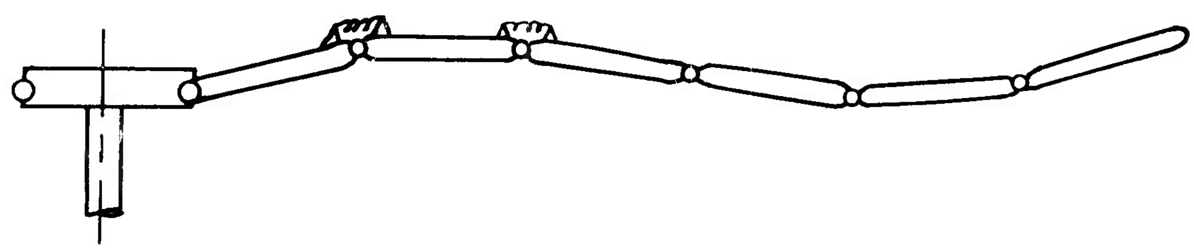


FIGURE 23. RANGE CHARACTERISTICS AT 150 KNOTS

ACTUAL BLADE



MATHEMATICAL EQUIVALENT

FIGURE 24. THEORETICAL REPRESENTATION OF FLEXIBLE BLADE

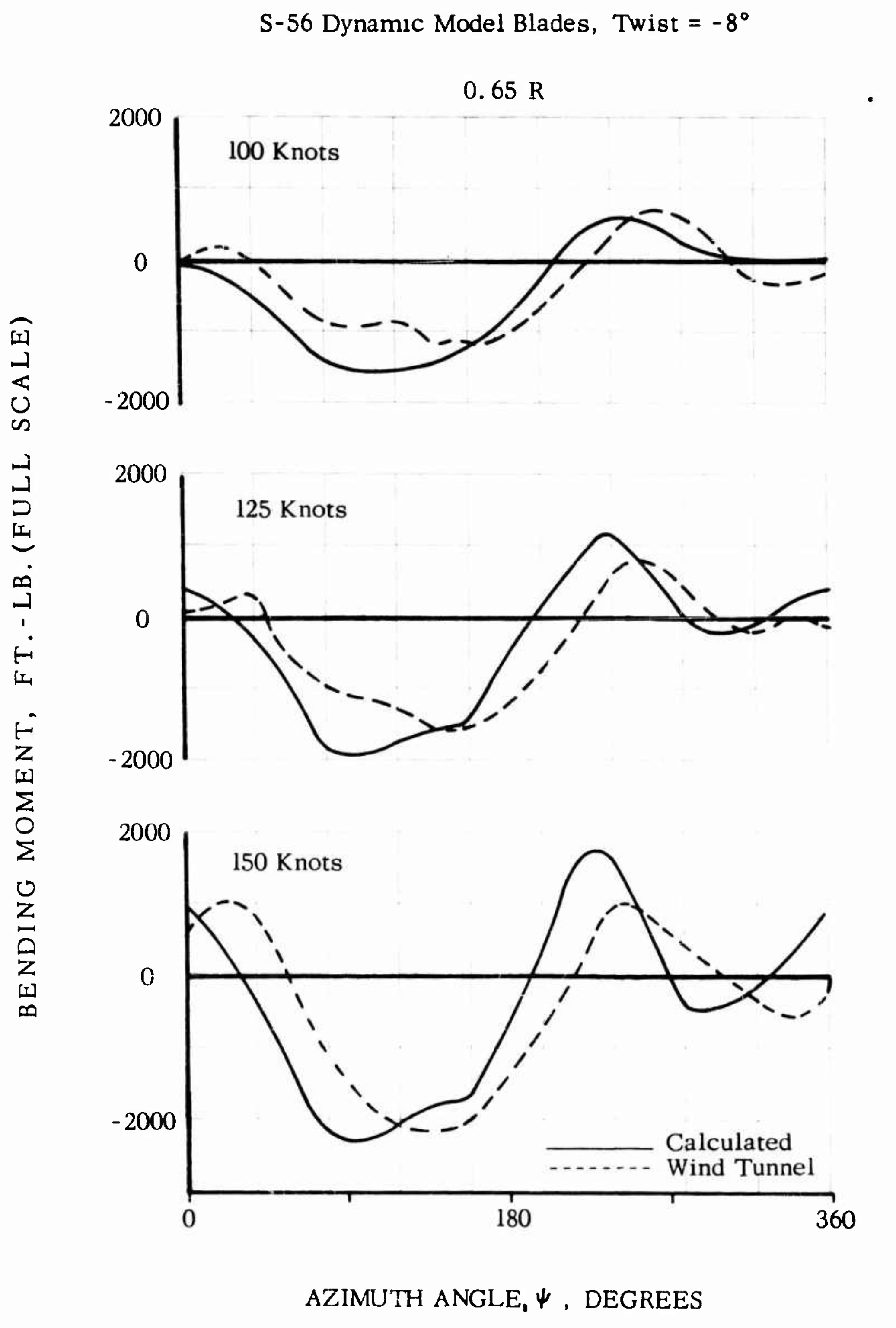


FIGURE 25. FLATWISE BENDING TIME HISTORIES, 0.65 R

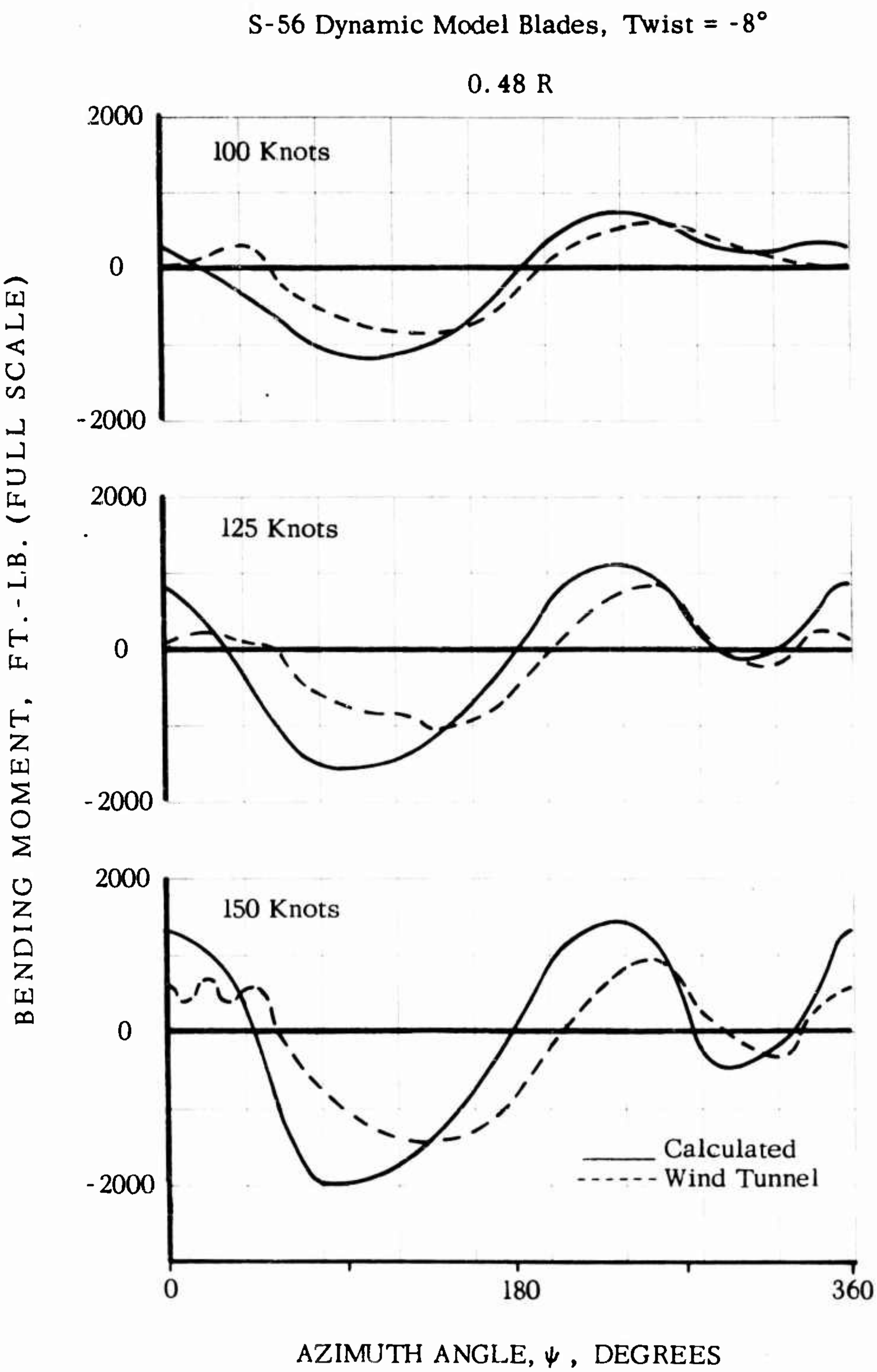


FIGURE 26. FLATWISE BENDING TIME HISTORIES, 0.48 R

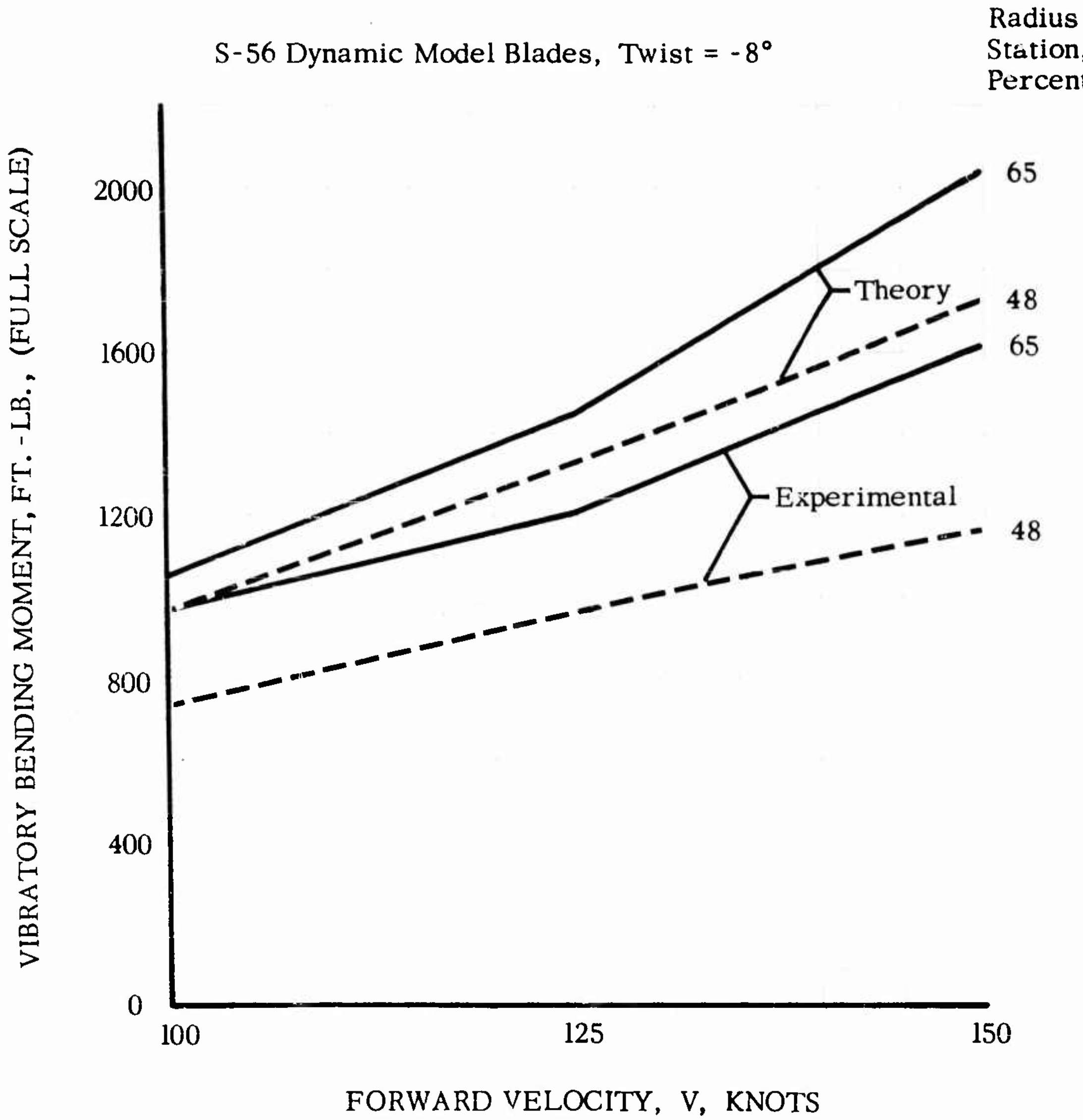


FIGURE 27. BENDING MOMENT AMPLITUDES

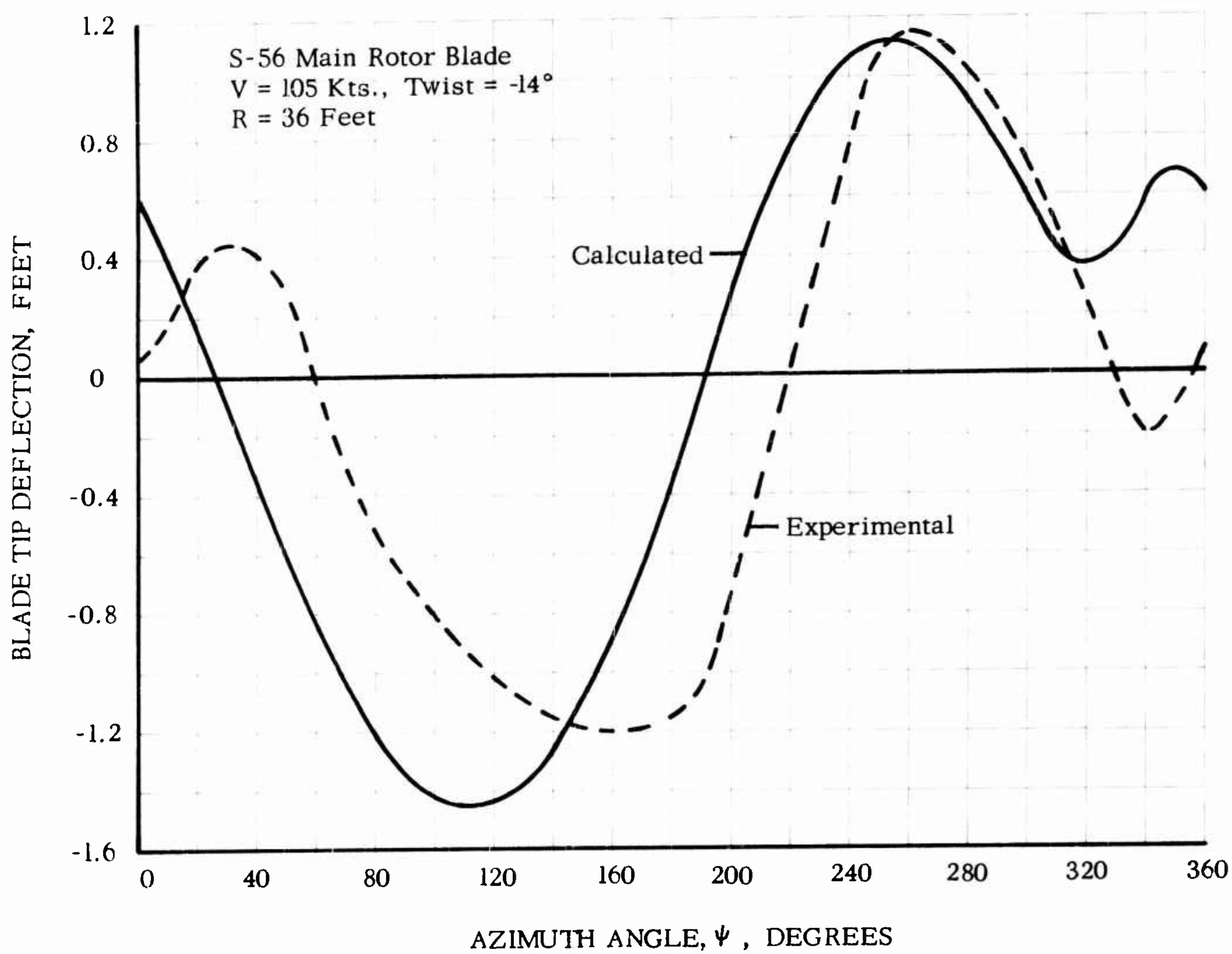


FIGURE 28. BLADE TIP DEFLECTION DUE TO FLATWISE BENDING

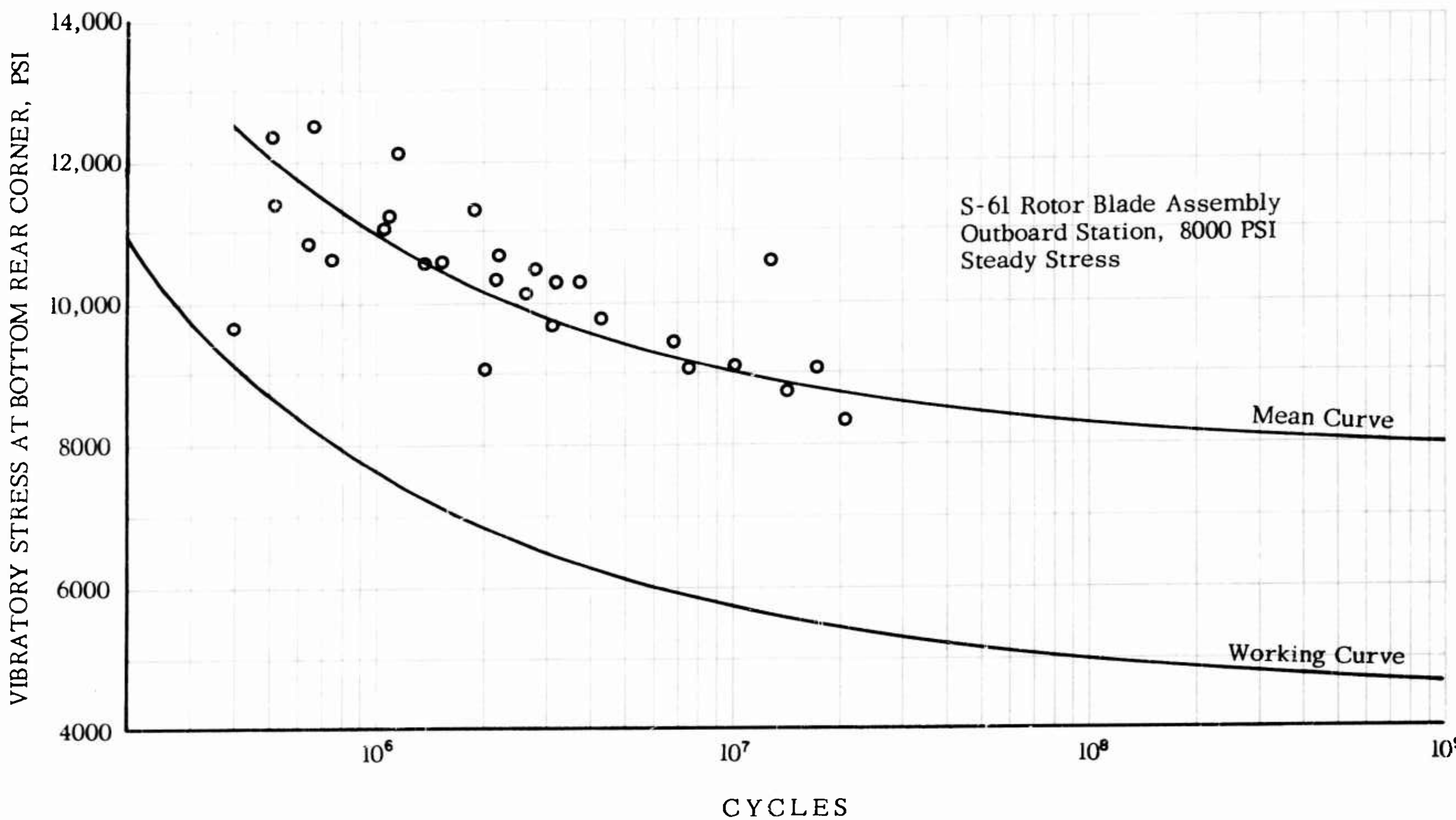


FIGURE 29. ALLOWABLE SPAR VIBRATORY STRESS

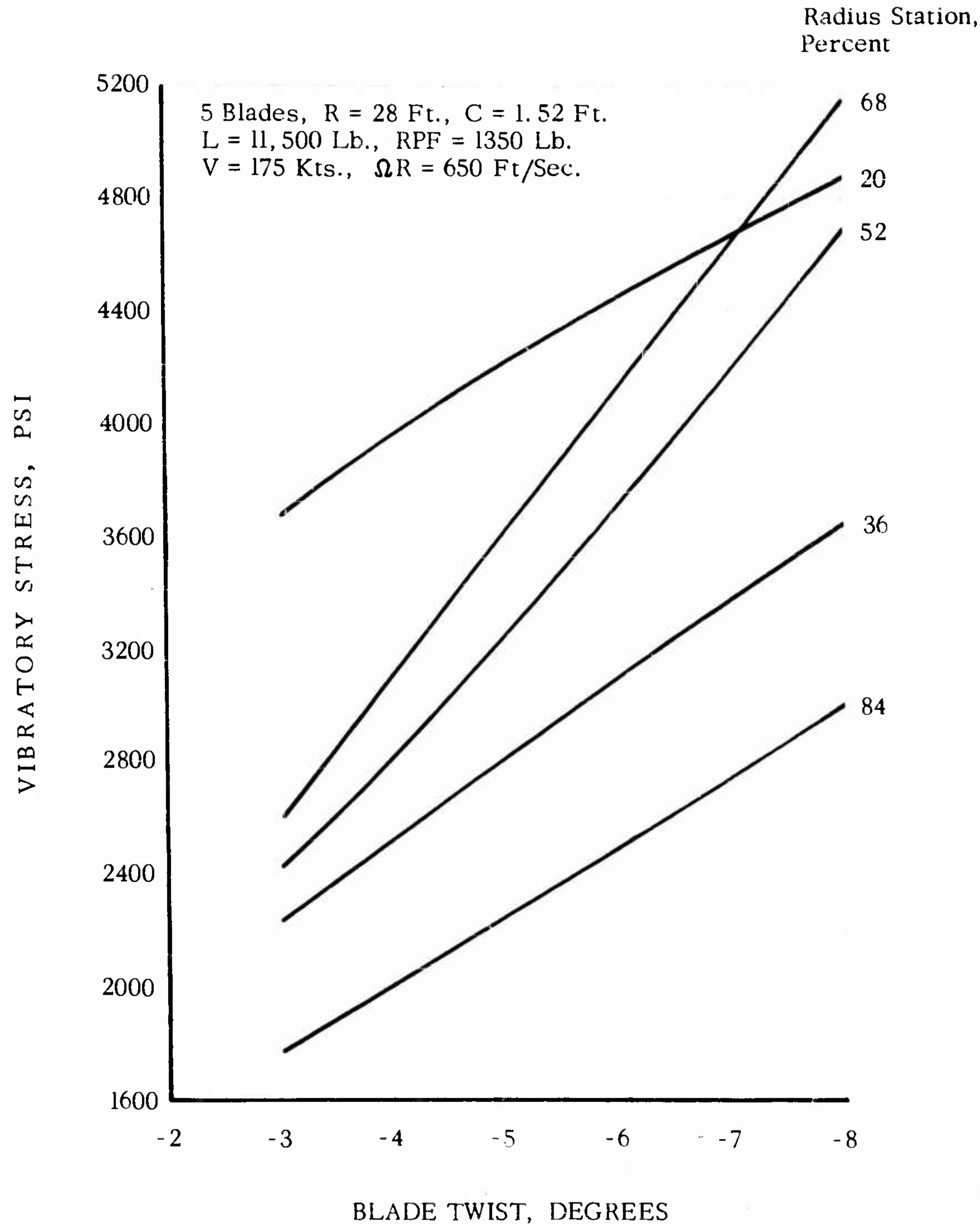


FIGURE 30. EFFECTS OF BLADE TWIST ON FLATWISE BENDING

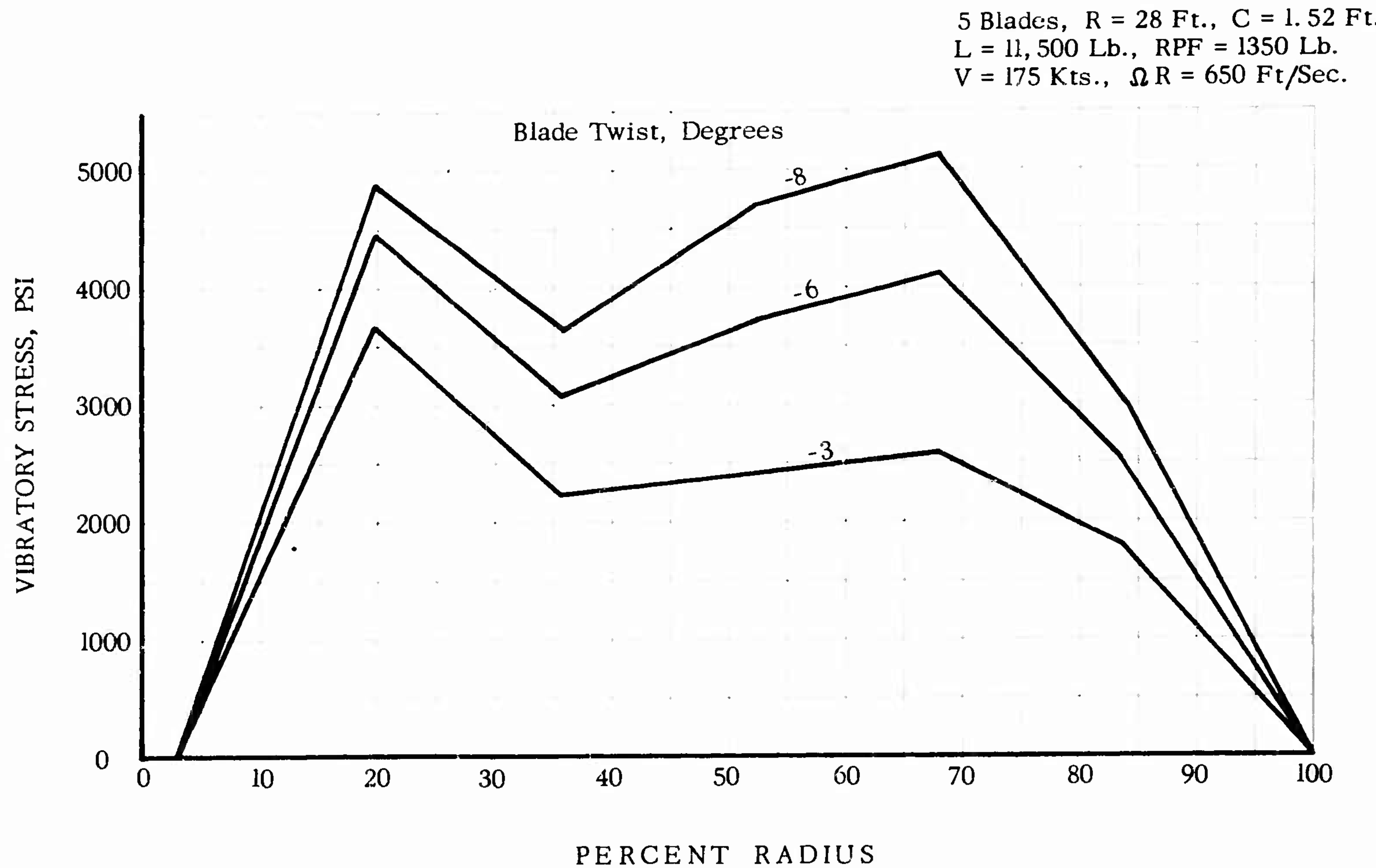


FIGURE 31. EFFECTS OF TWIST ON STRESS DISTRIBUTION

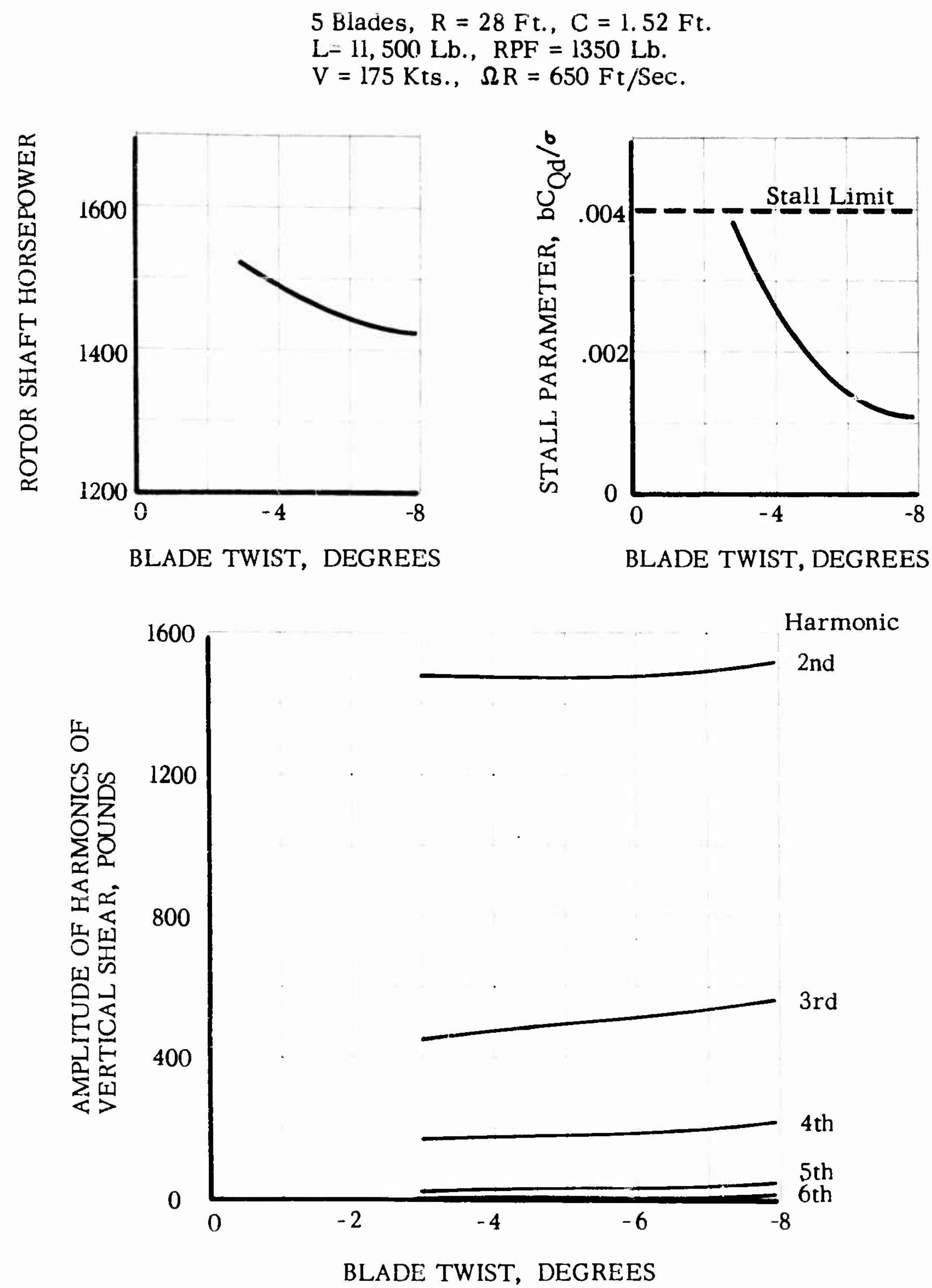


FIGURE 32. ADDITIONAL EFFECTS OF TWIST

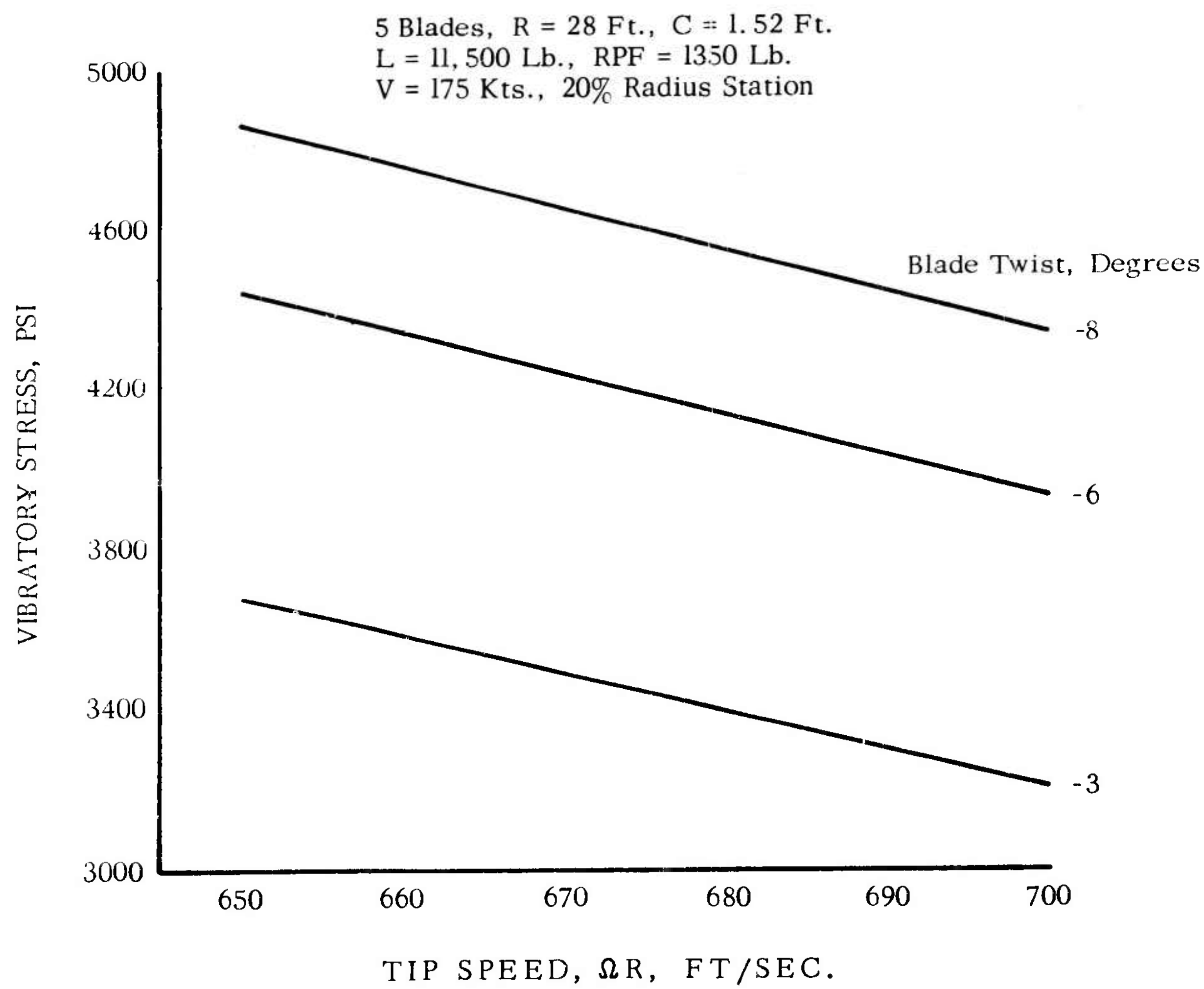


FIGURE 33. EFFECT OF TIP SPEED ON VIBRATORY STRESS

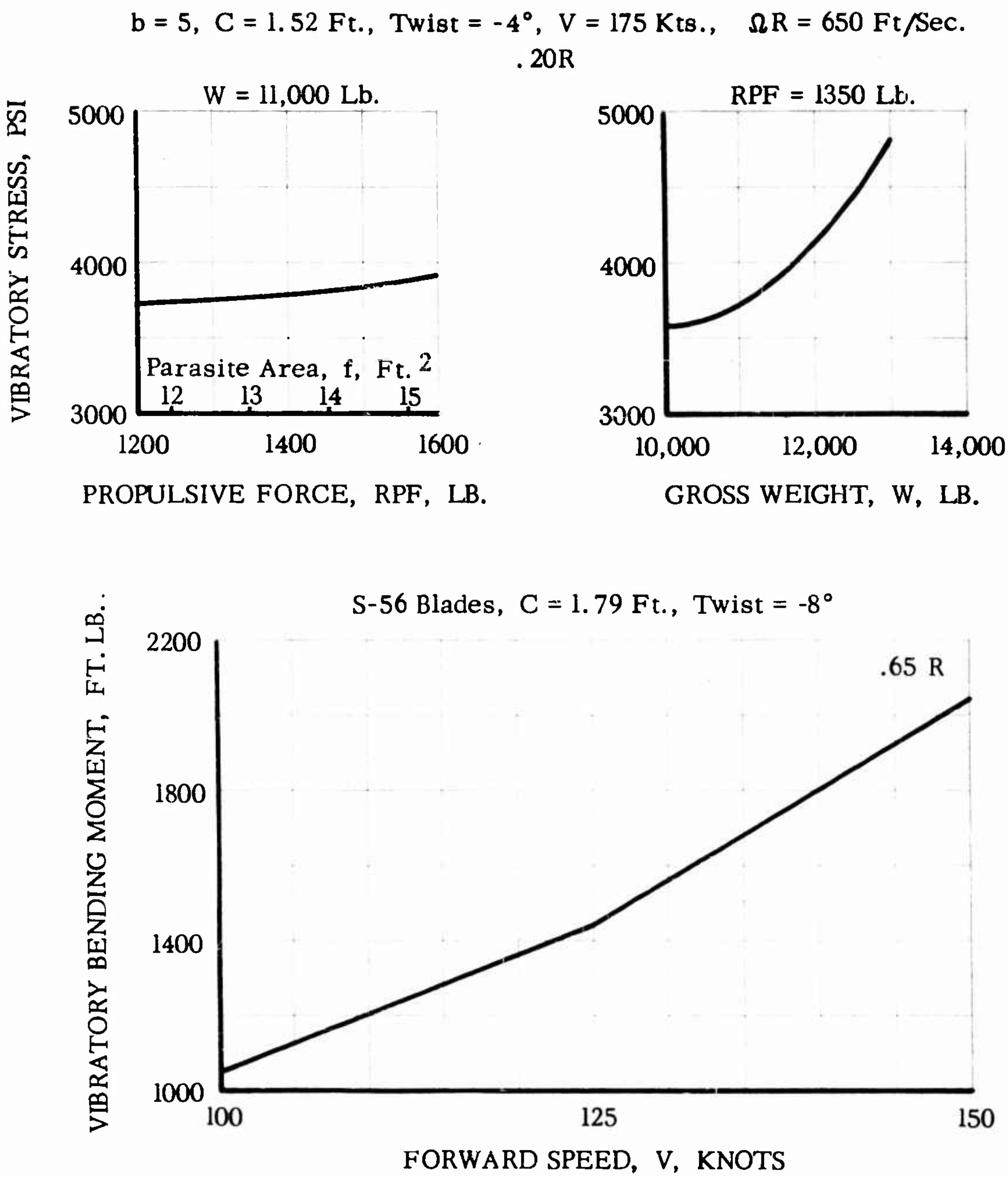


FIGURE 34. EFFECTS OF OTHER VARIABLES ON VIBRATORY STRESS

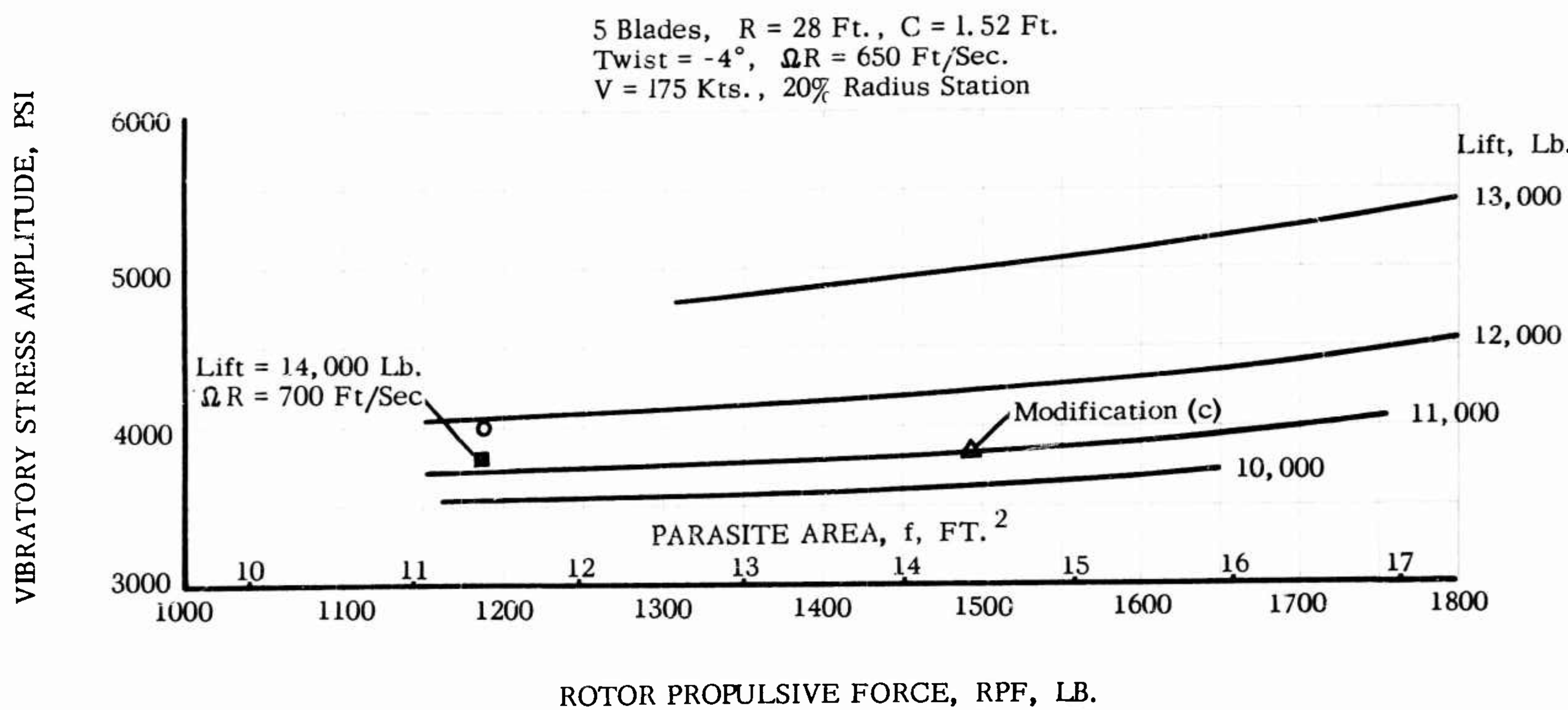


FIGURE 35. VIBRATORY STRESS FOR SELECTED ROTOR

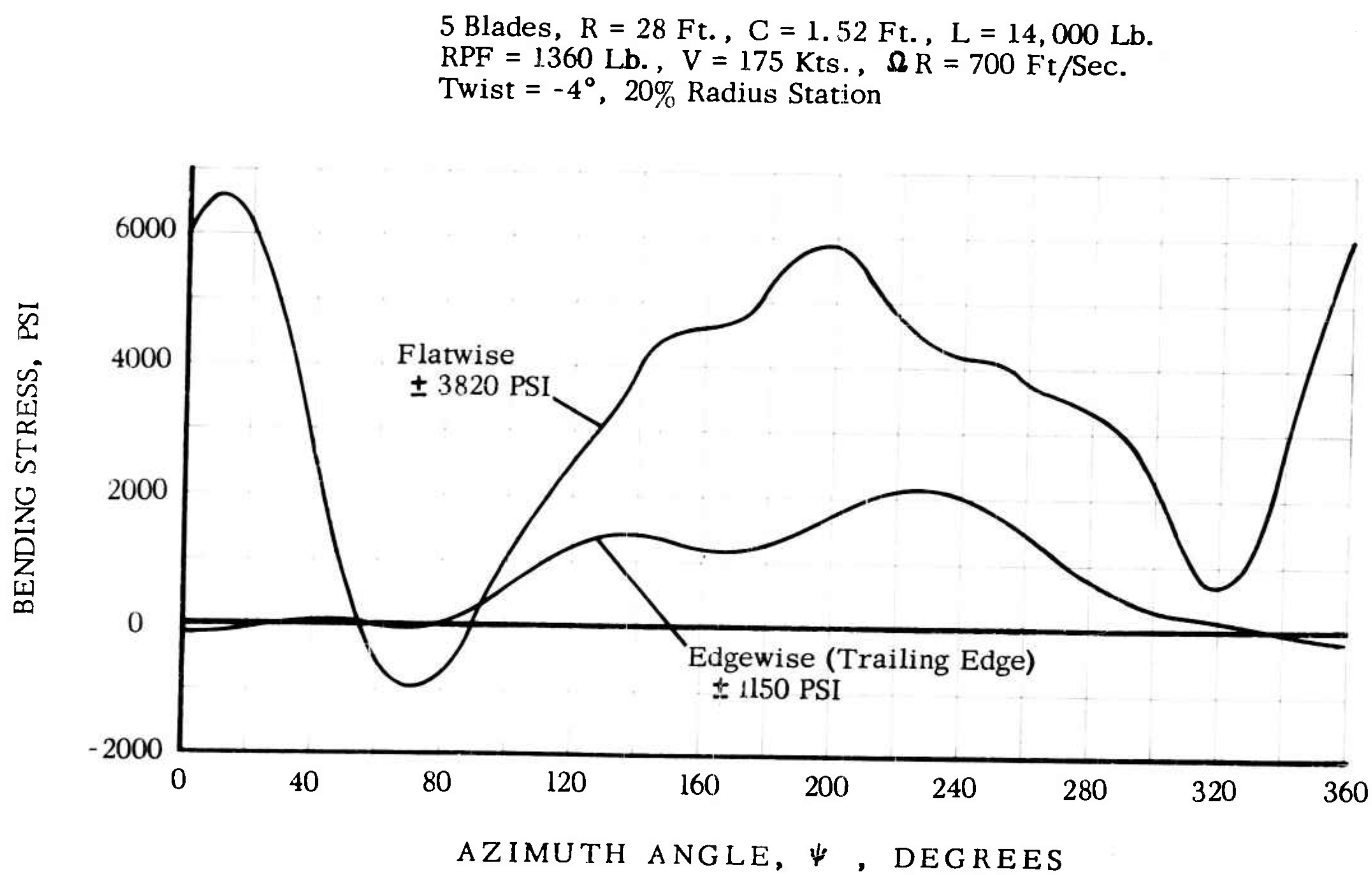


FIGURE 36. COMPARISON OF FLATWISE AND EDGEWISE STRESSES

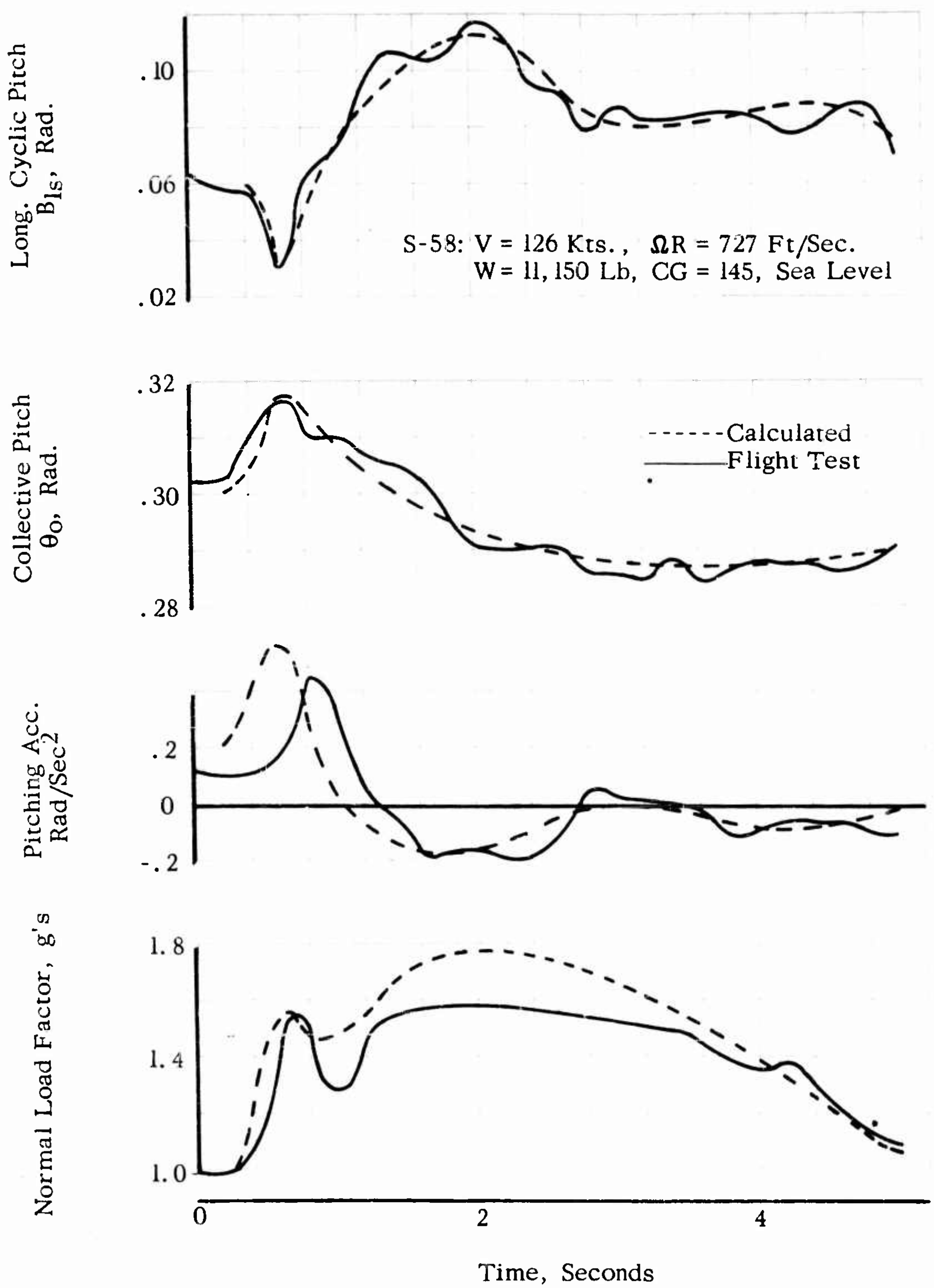


FIGURE 37. TIME HISTORY OF SYMMETRICAL PULLOUT MANEUVER

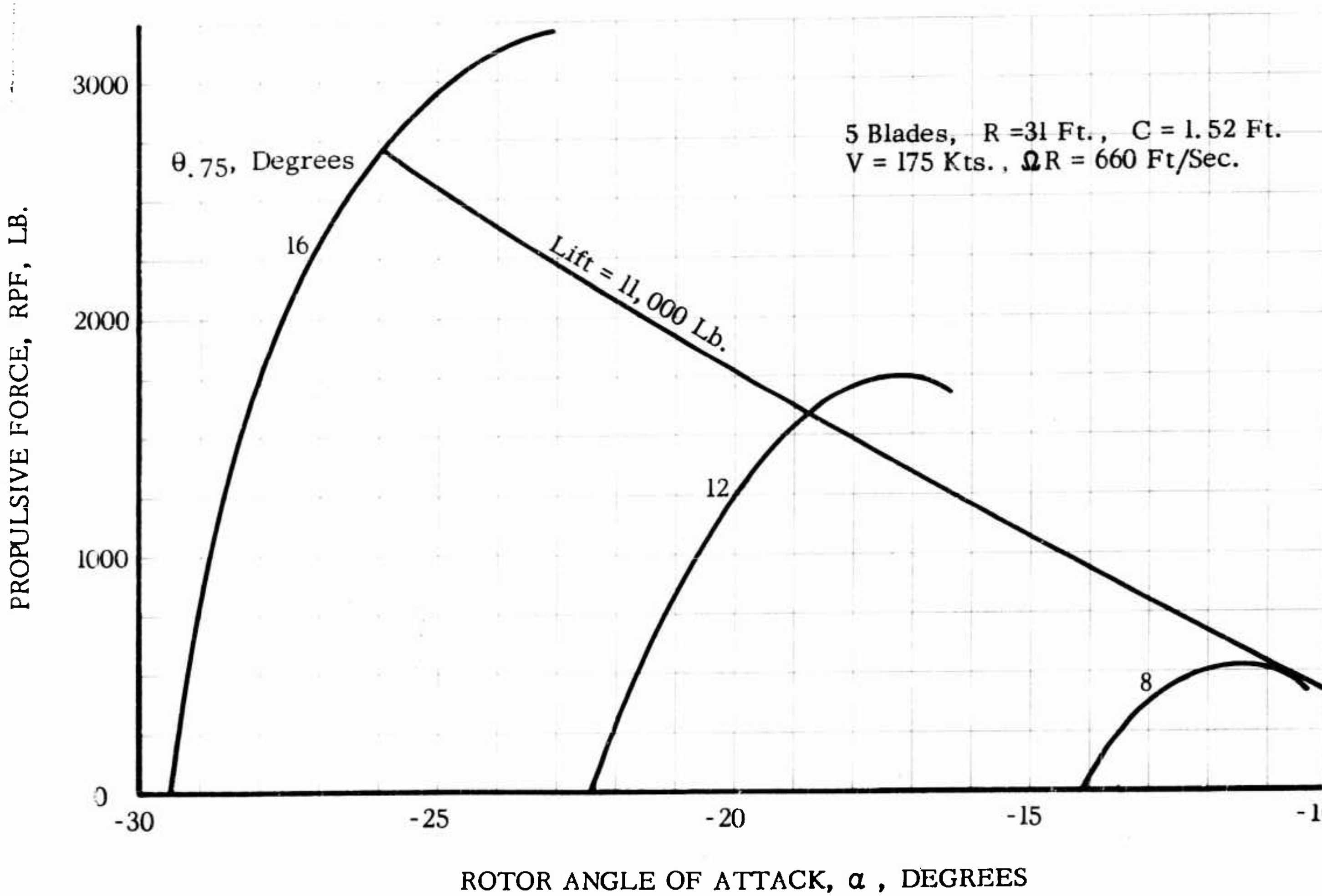


FIGURE 38. VARIATION OF PROPULSIVE FORCE WITH ANGLE OF ATTACK

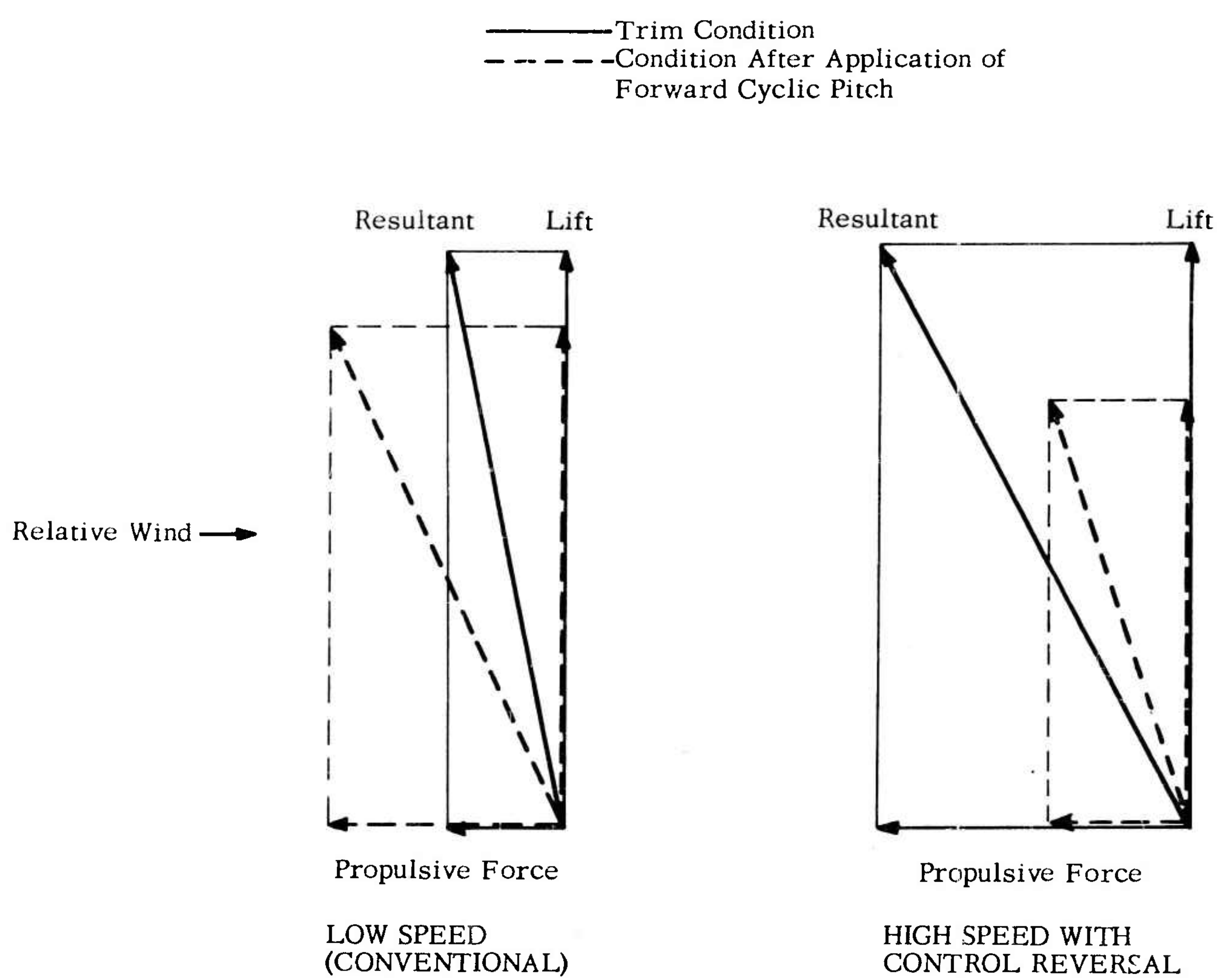


FIGURE 39. EFFECT OF CYCLIC PITCH ON ROTOR RESULTANT FORCE

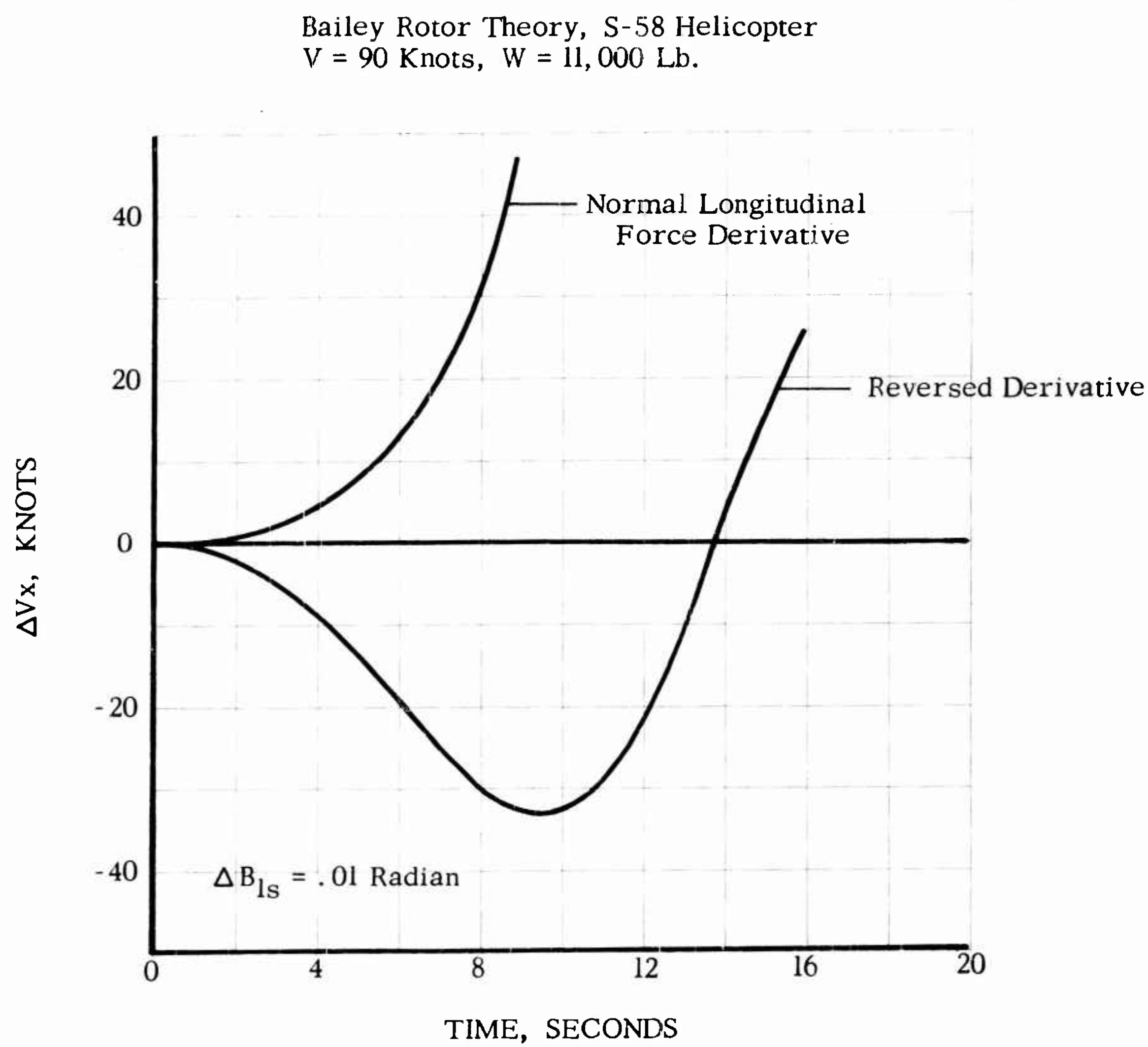


FIGURE 40. AIRSPEED RESPONSE WITH NORMAL AND REVERSED CONTROL

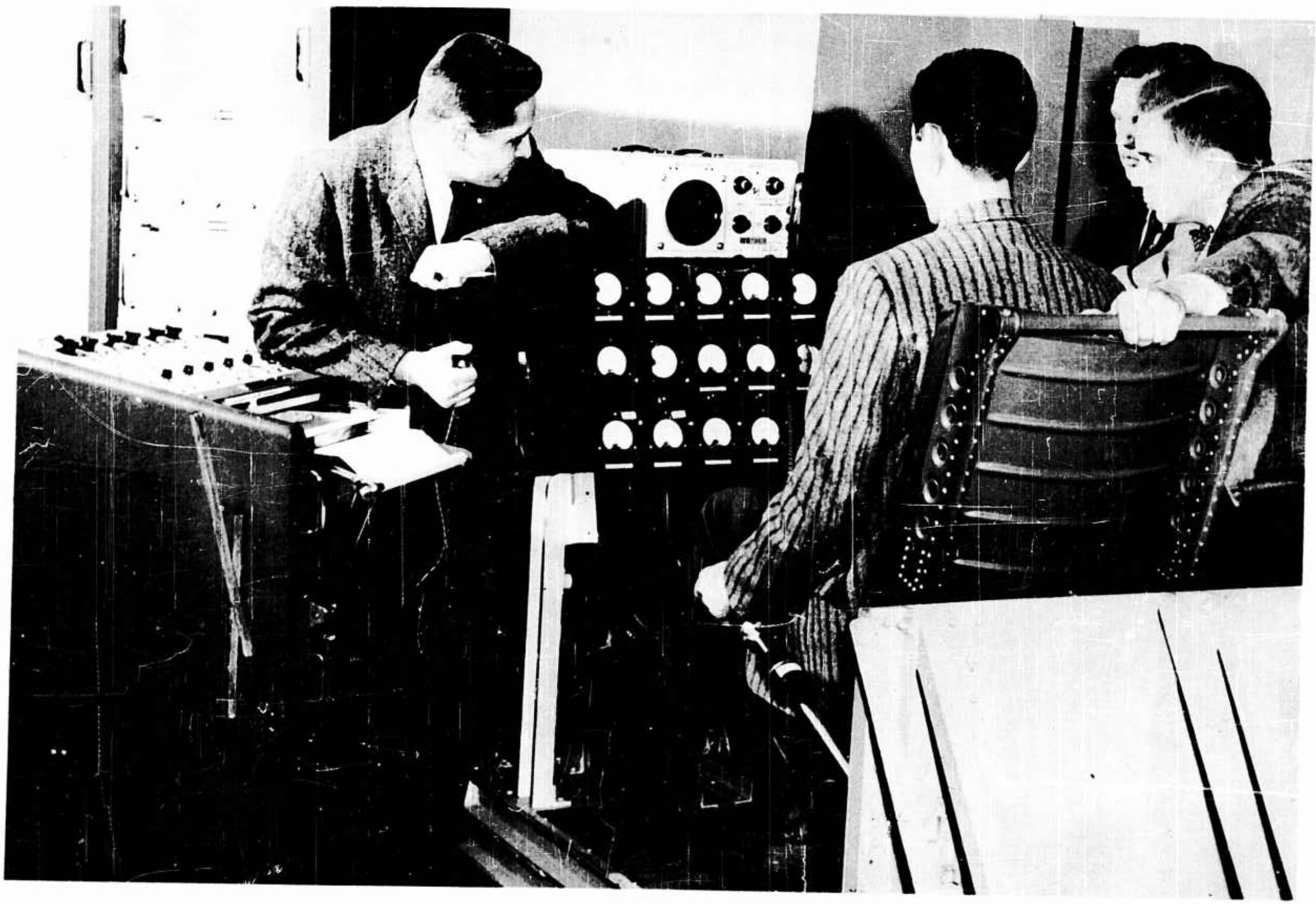


FIGURE 41. HELICOPTER FLIGHT SIMULATOR

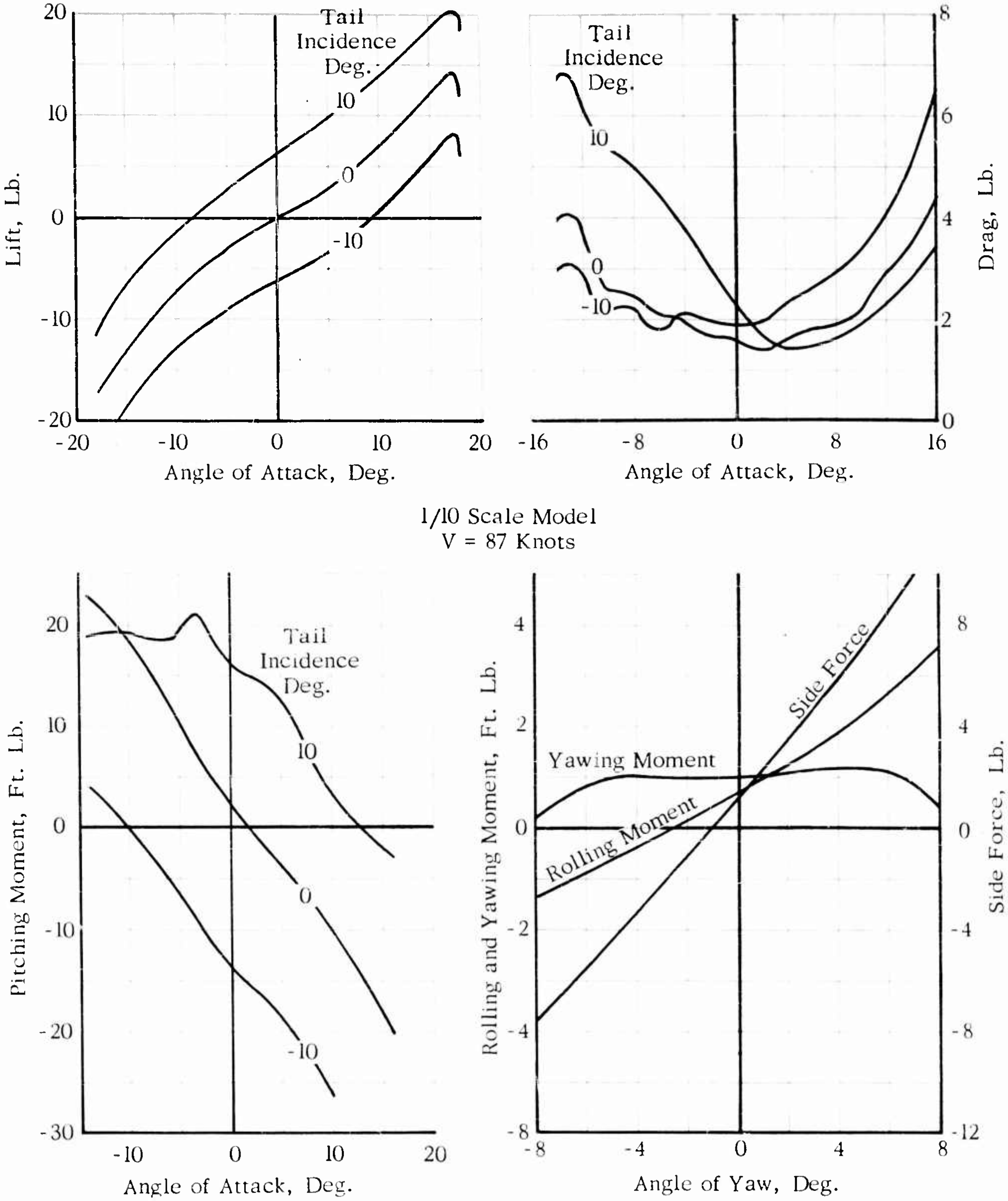


FIGURE 42. BODY AERODYNAMIC CHARACTERISTICS USED IN SIMULATOR STUDY

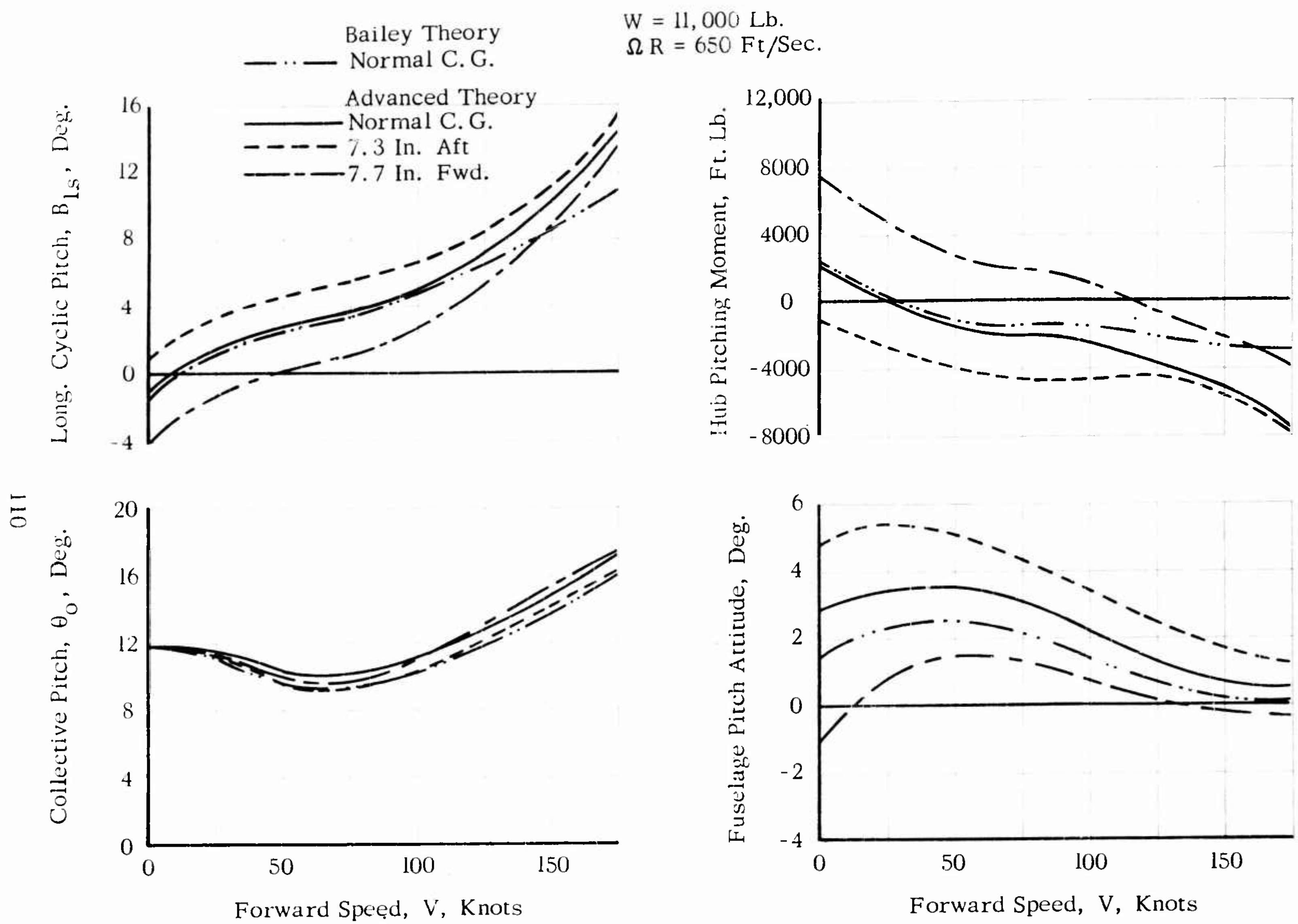


FIGURE 43. STATIC TRIM FOR HIGH PERFORMANCE HELICOPTER, W = 11,000 POUNDS

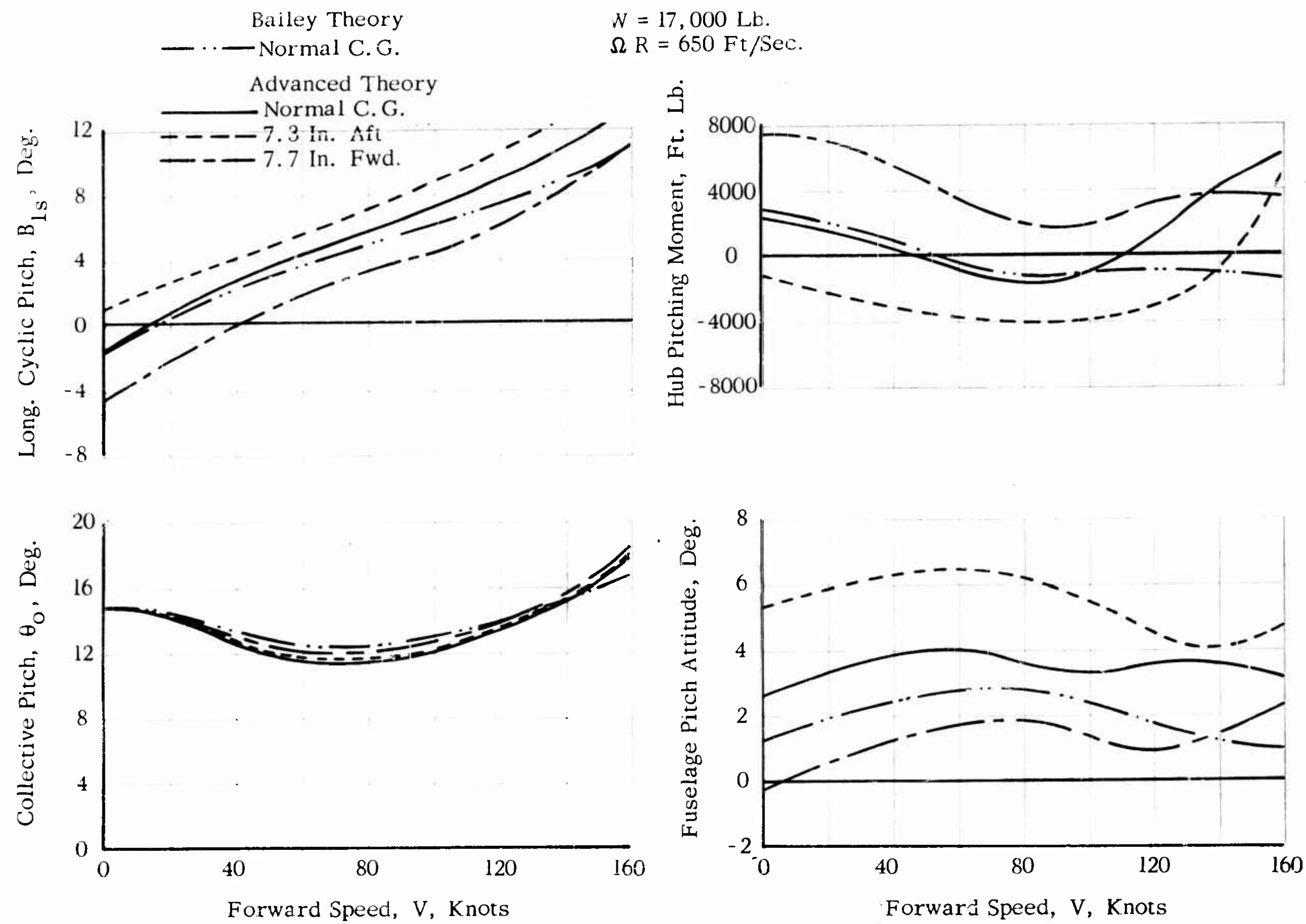


FIGURE 44. STATIC TRIM FOR HIGH PERFORMANCE HELICOPTER, W = 17,000 POUNDS

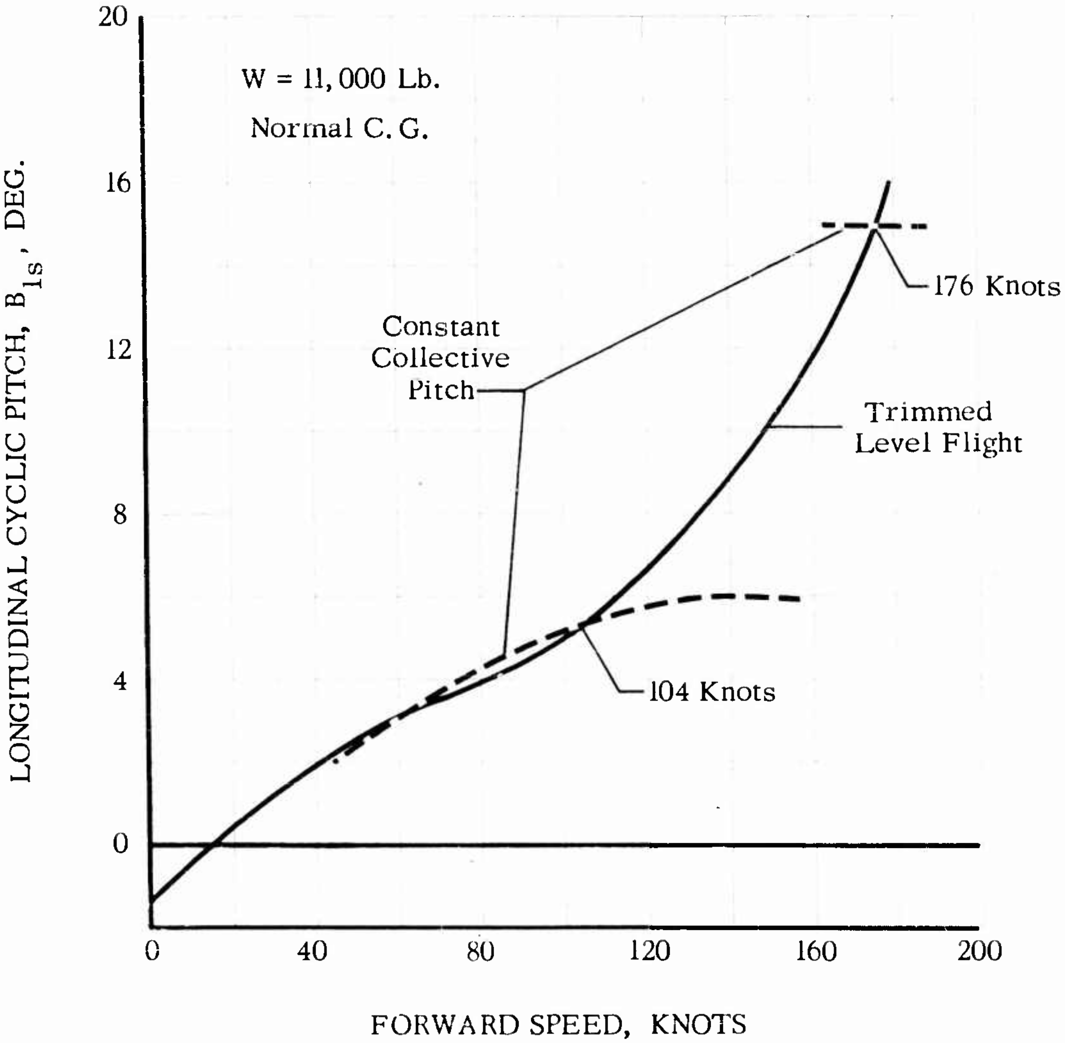


FIGURE 45. LONGITUDINAL CYCLIC CONTROL POSITION STABILITY

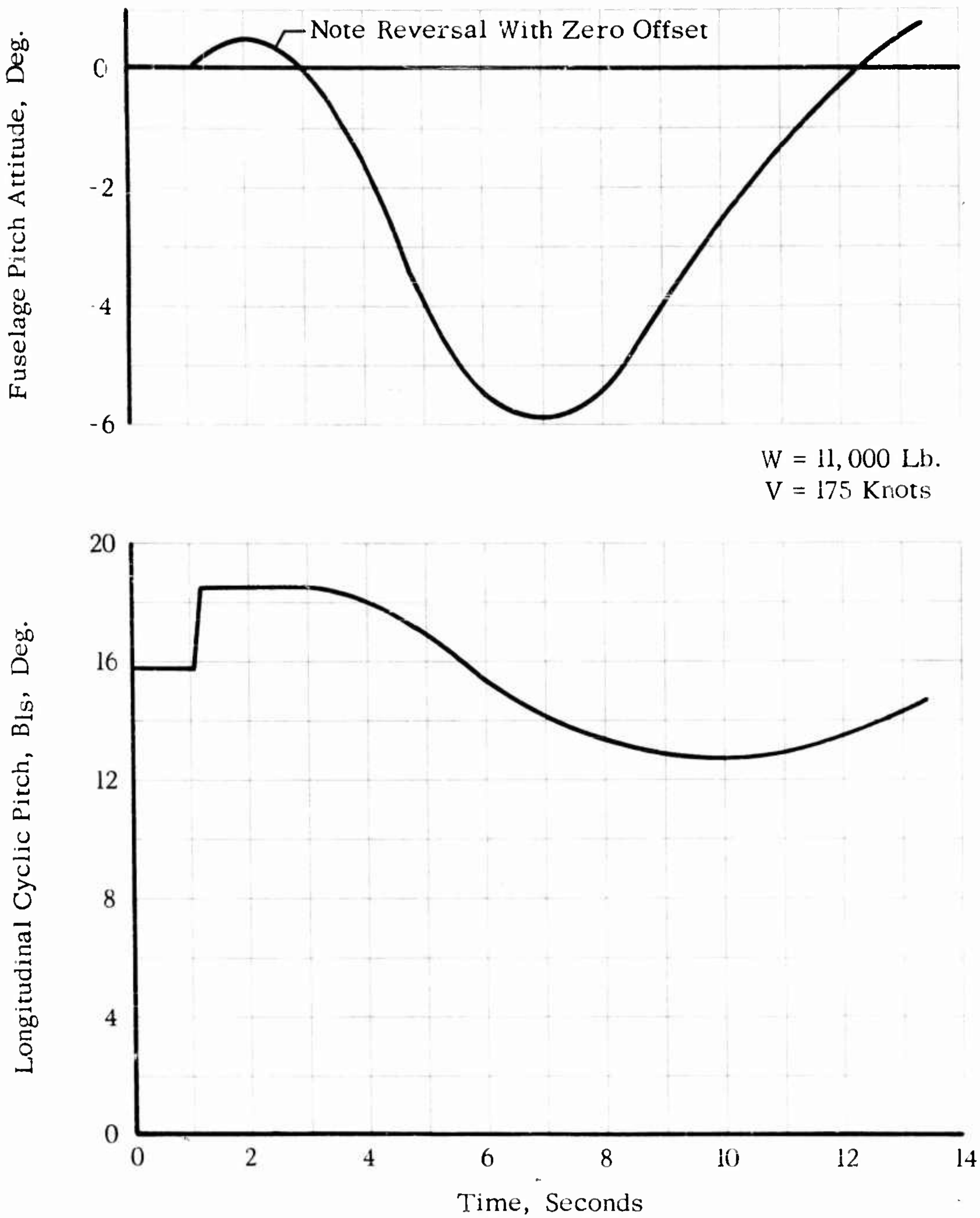


FIGURE 46. LONGITUDINAL CONTROL RESPONSE WITH ZERO FLAPPING HINGE OFFSET

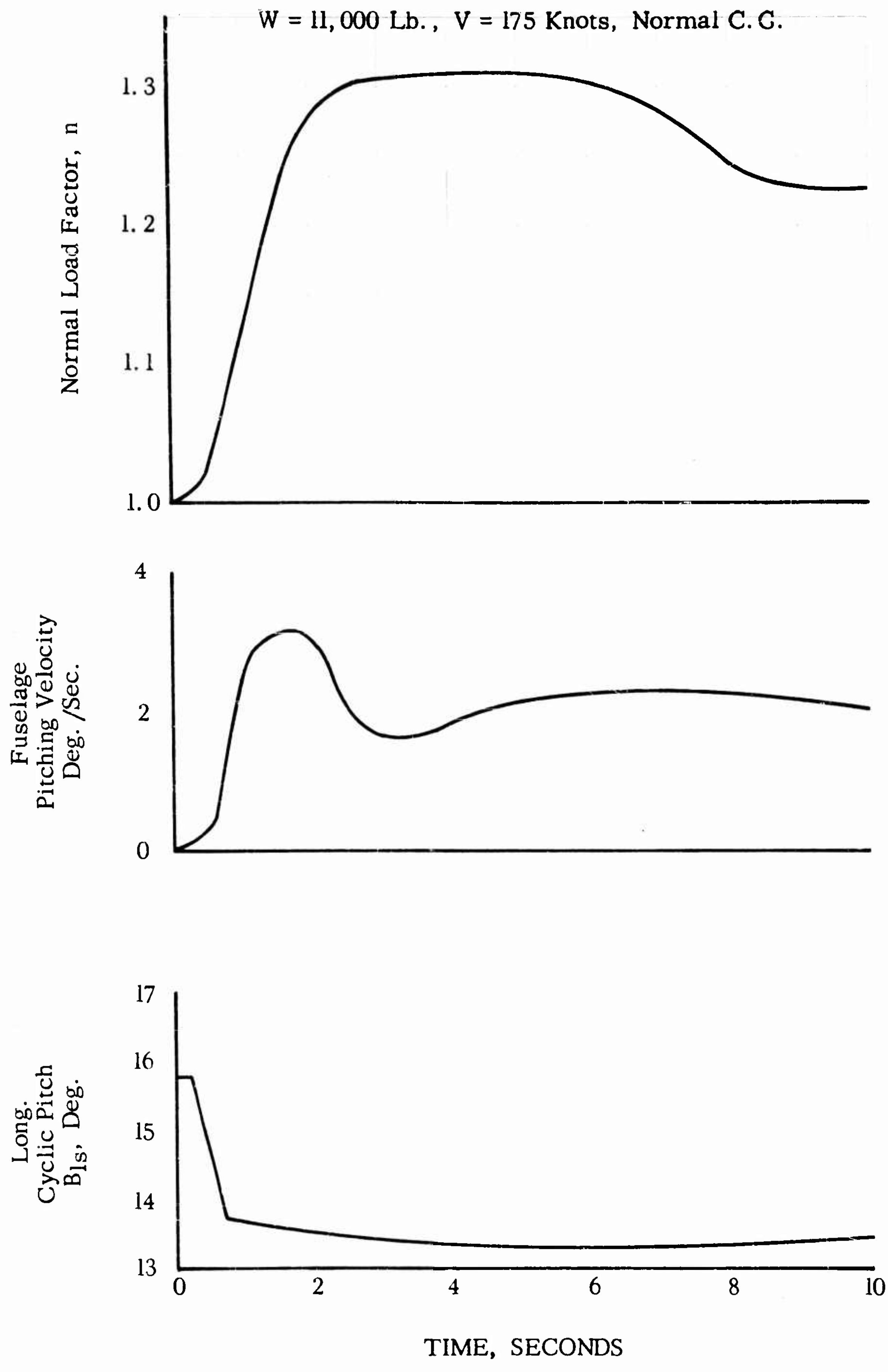


FIGURE 47. HPH LONGITUDINAL MANEUVERING RESPONSE, 175 KNOTS

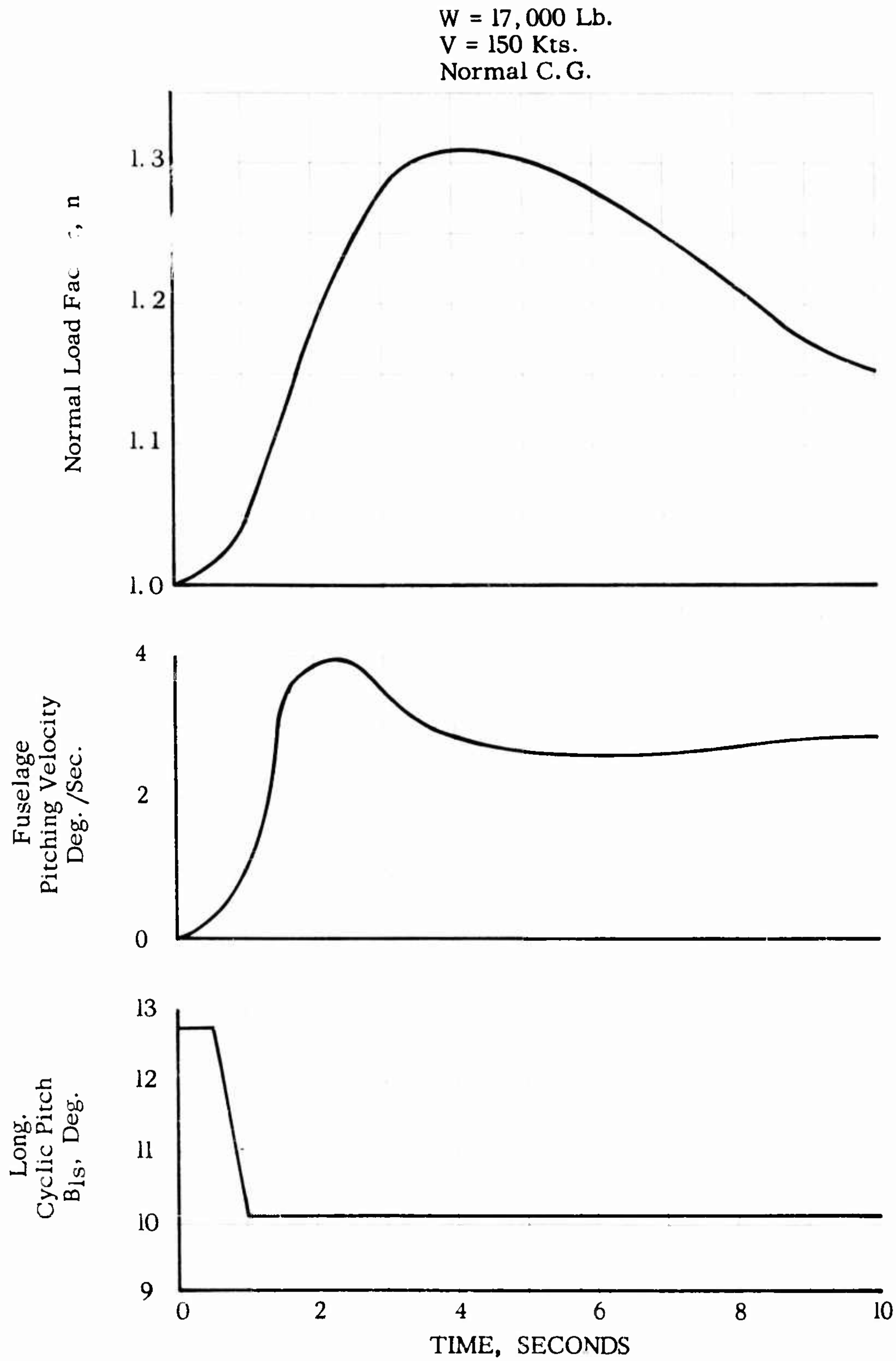


FIGURE 48. HPH LONGITUDINAL MANEUVERING RESPONSE, 150 KNOTS

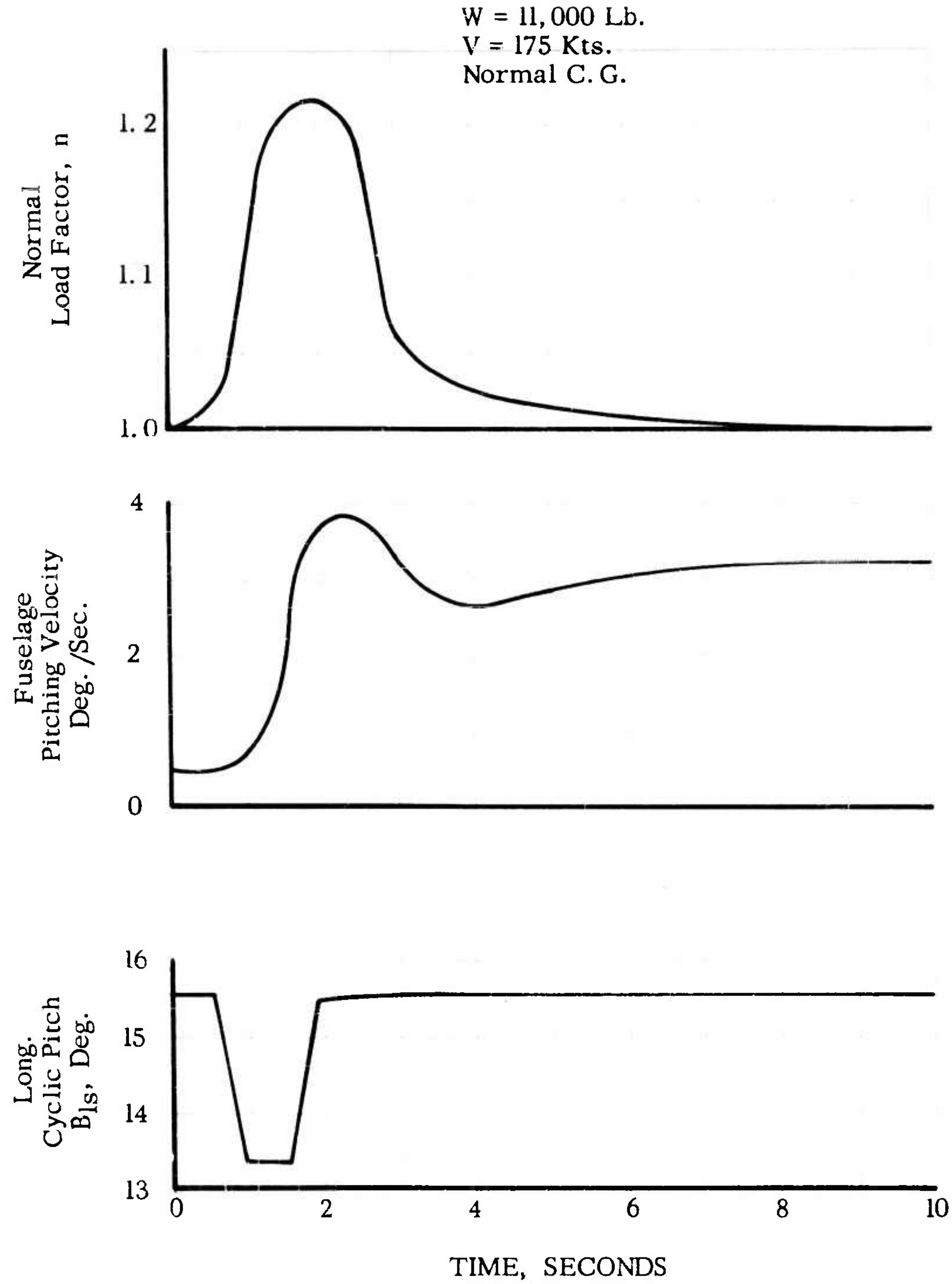


FIGURE 49. HPH LONGITUDINAL DYNAMIC STABILITY CHARACTERISTICS, 175 KNOTS

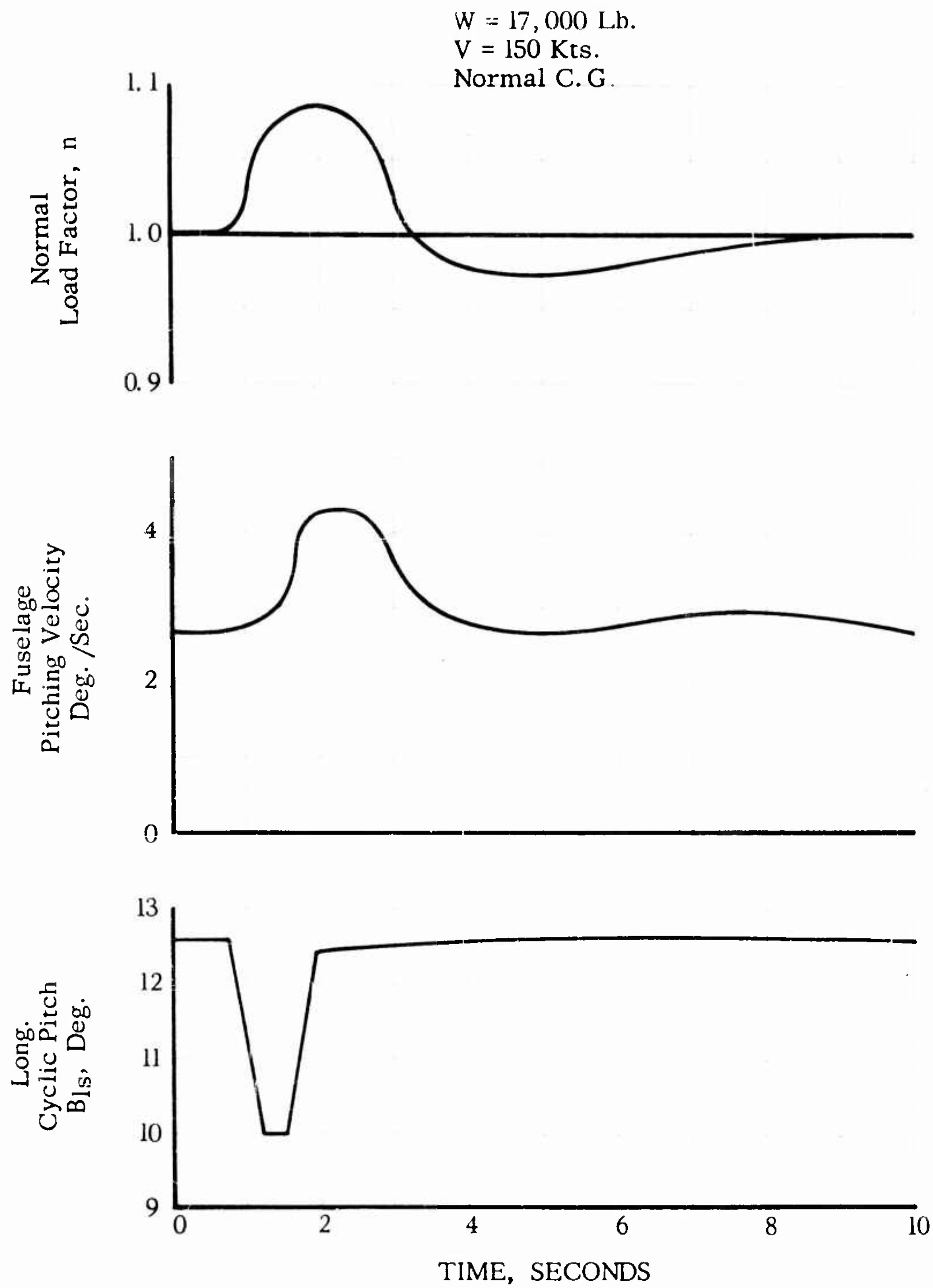


FIGURE 50. LONGITUDINAL DYNAMIC STABILITY CHARACTERISTICS, 150 KNOTS

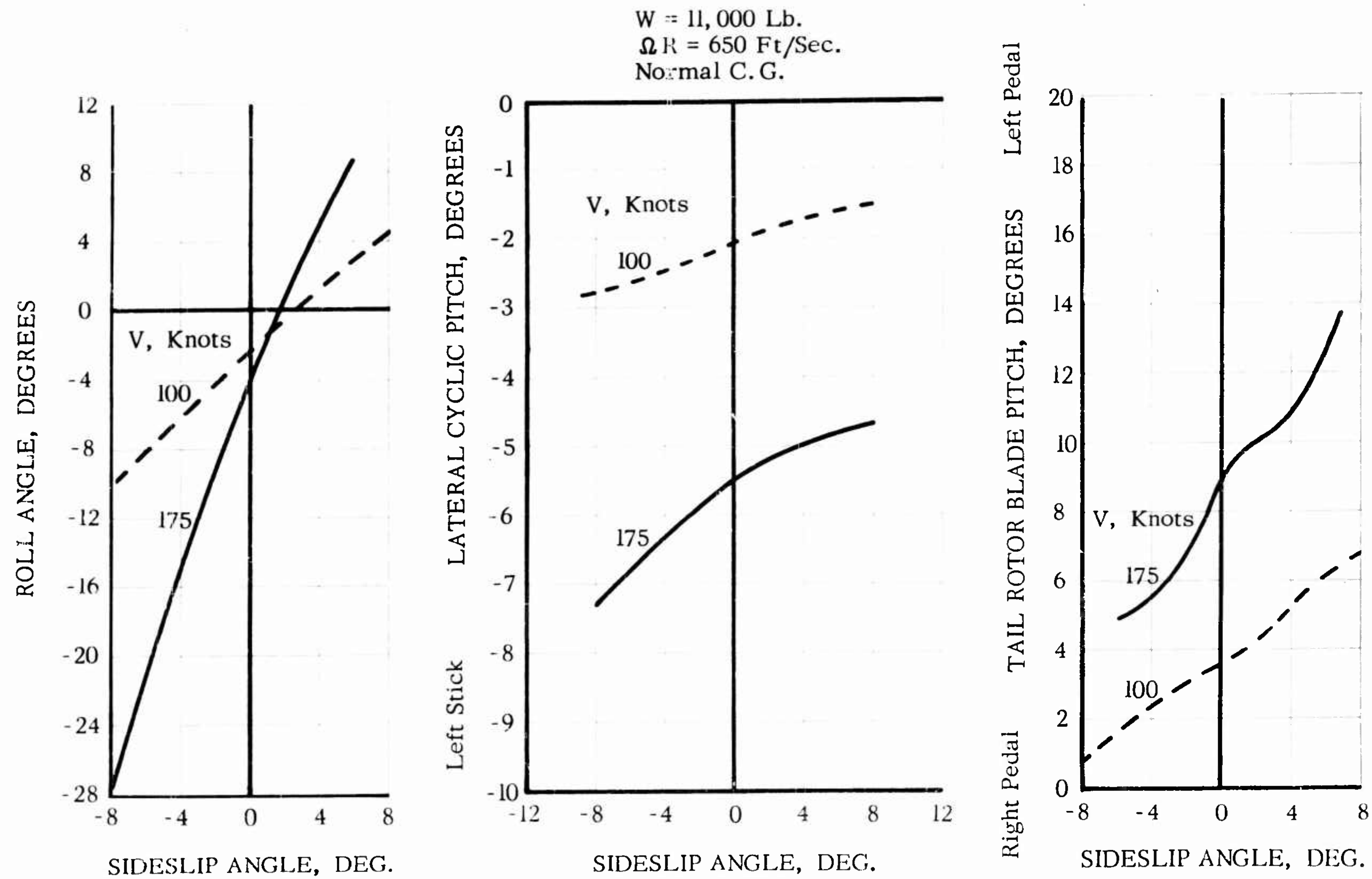


FIGURE 51. HPH LATERAL STATIC TRIM CHARACTERISTICS, W = 11,000 LB.

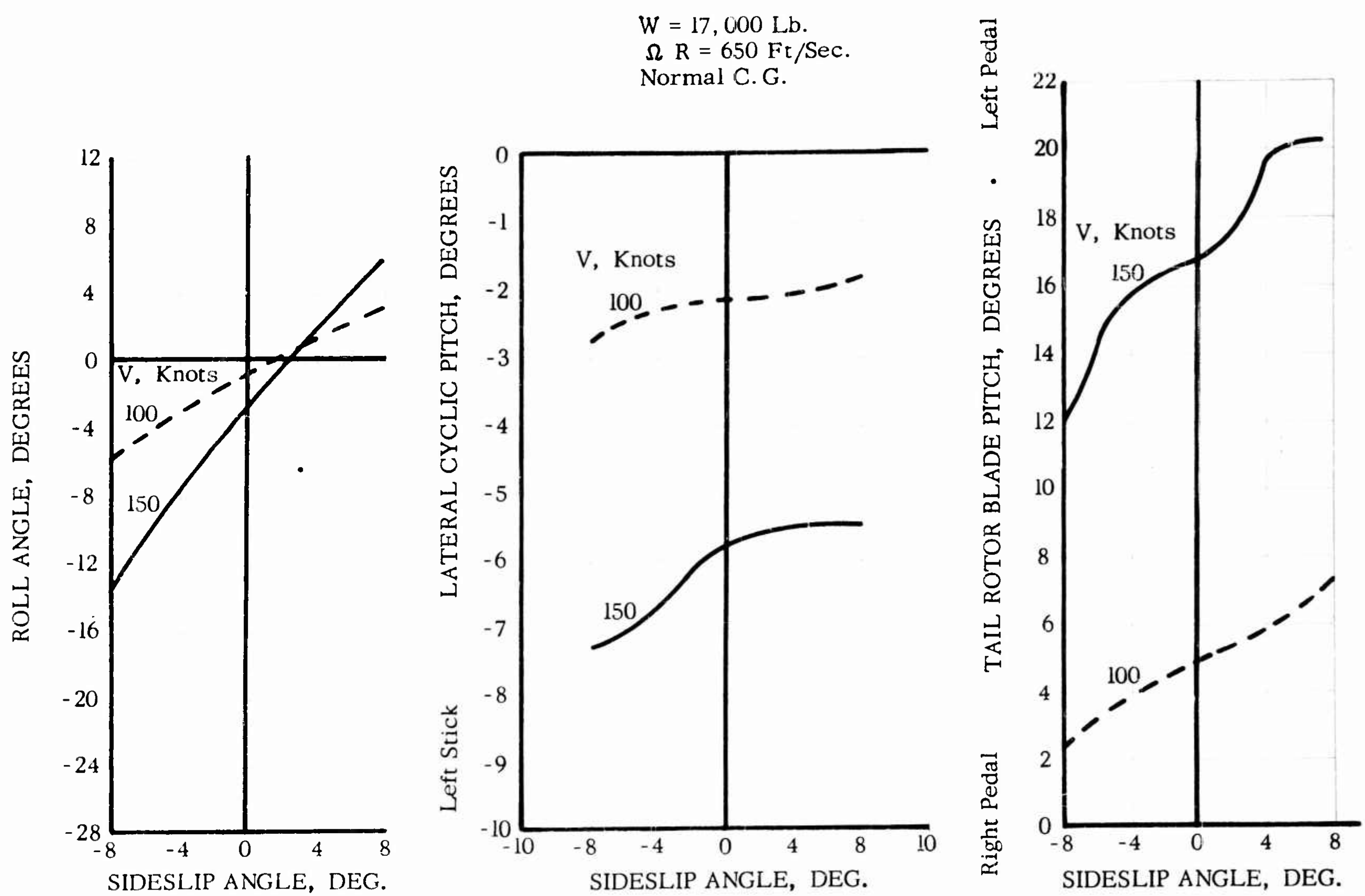


FIGURE 52. HPH LATERAL STATIC TRIM CHARACTERISTICS, W ≐ 17,000 LB.

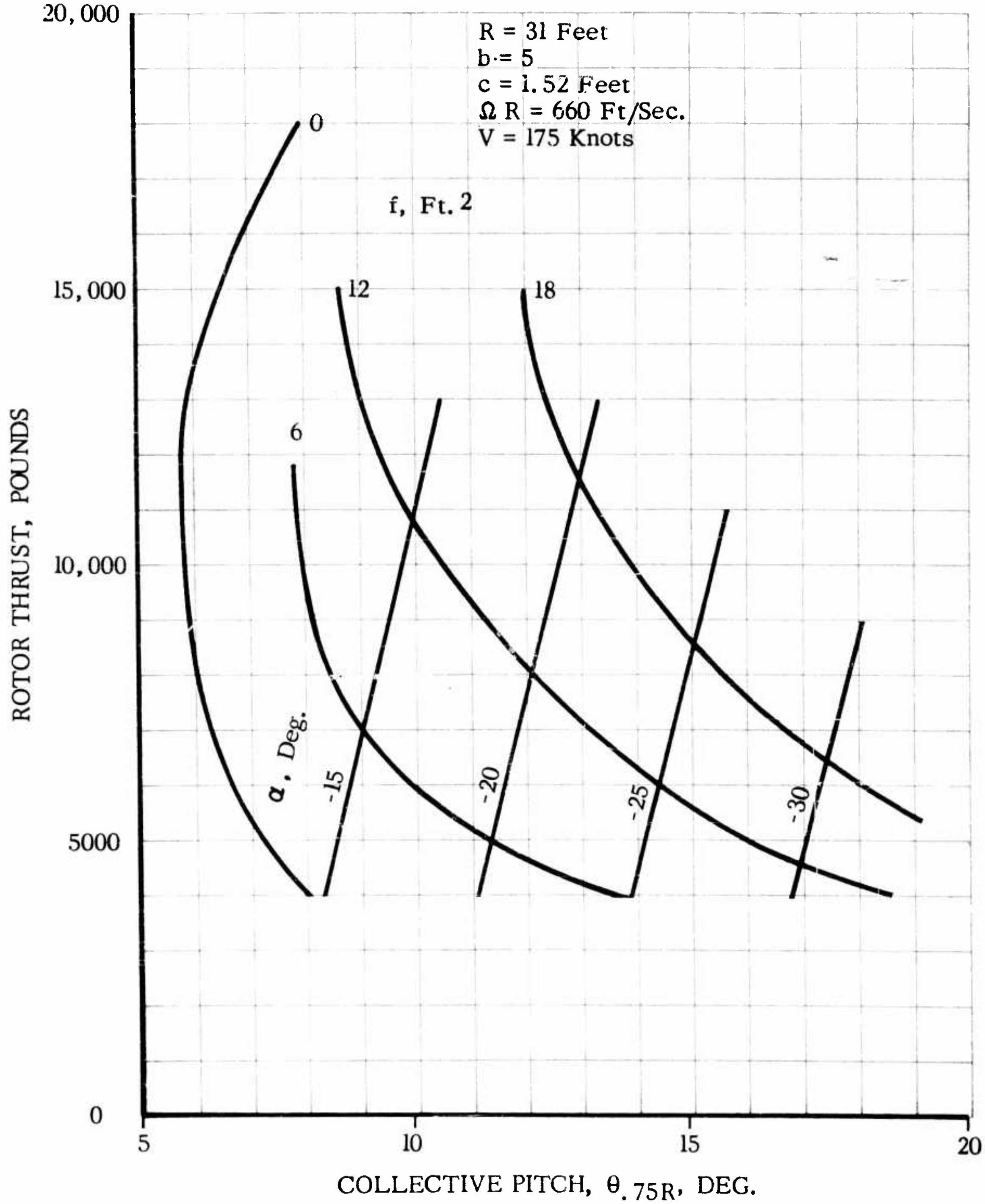


FIGURE 53. VARIATION OF THRUST WITH COLLECTIVE PITCH

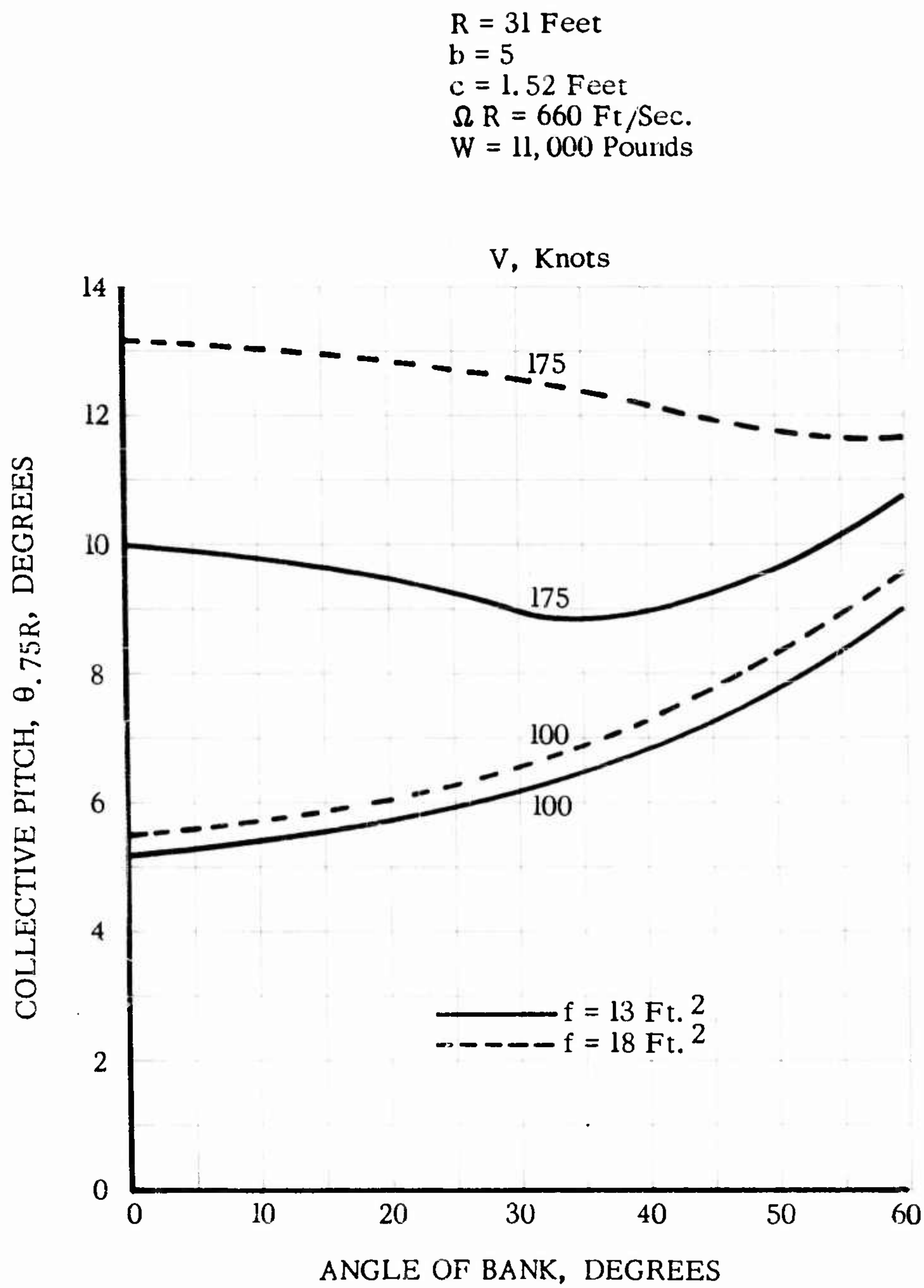


FIGURE 54. COLLECTIVE PITCH REQUIRED FOR COORDINATED TURNS

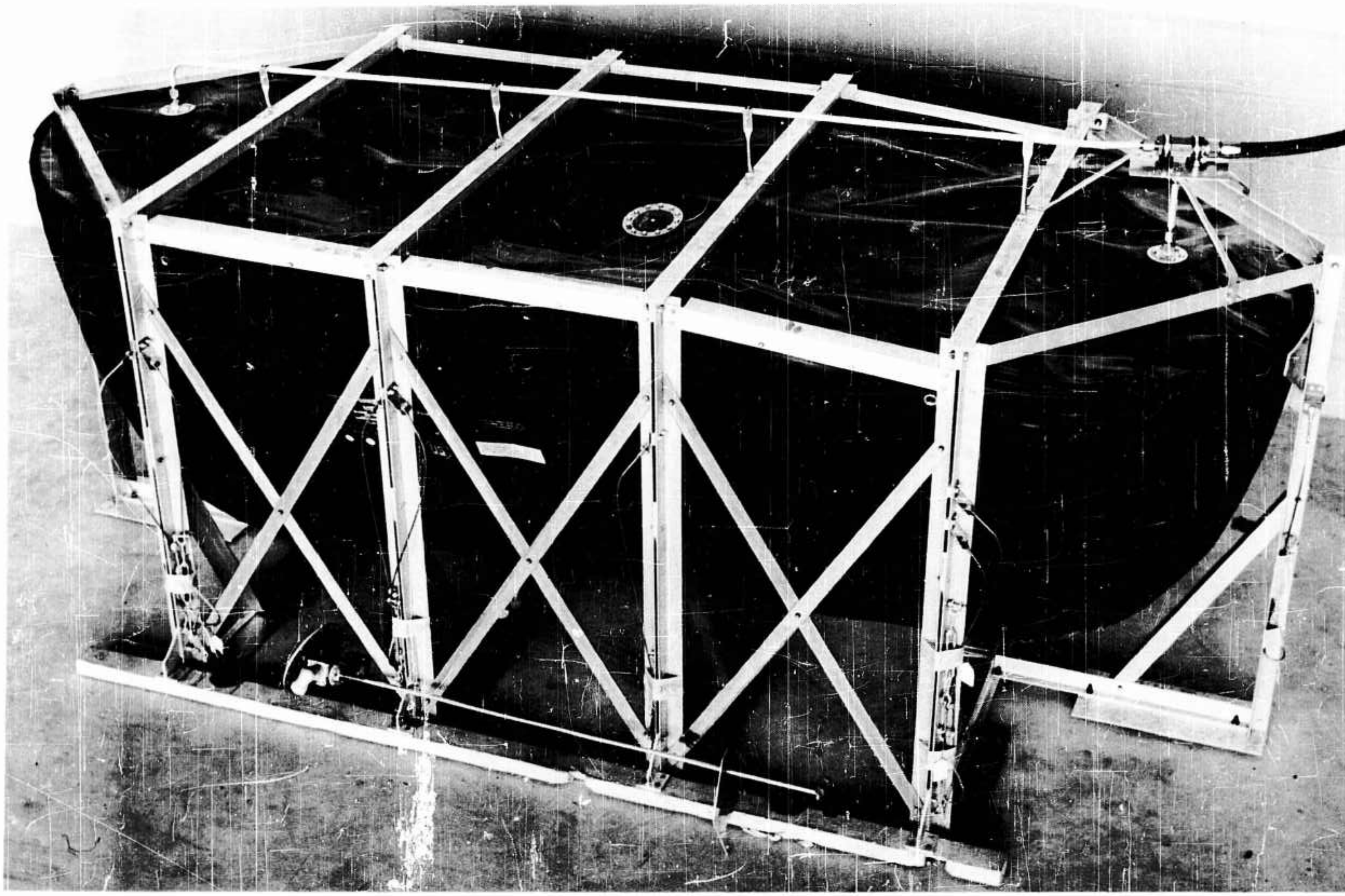


FIGURE 55. FERRY TANK (800 GAL.) USED IN S-58 HELICOPTER

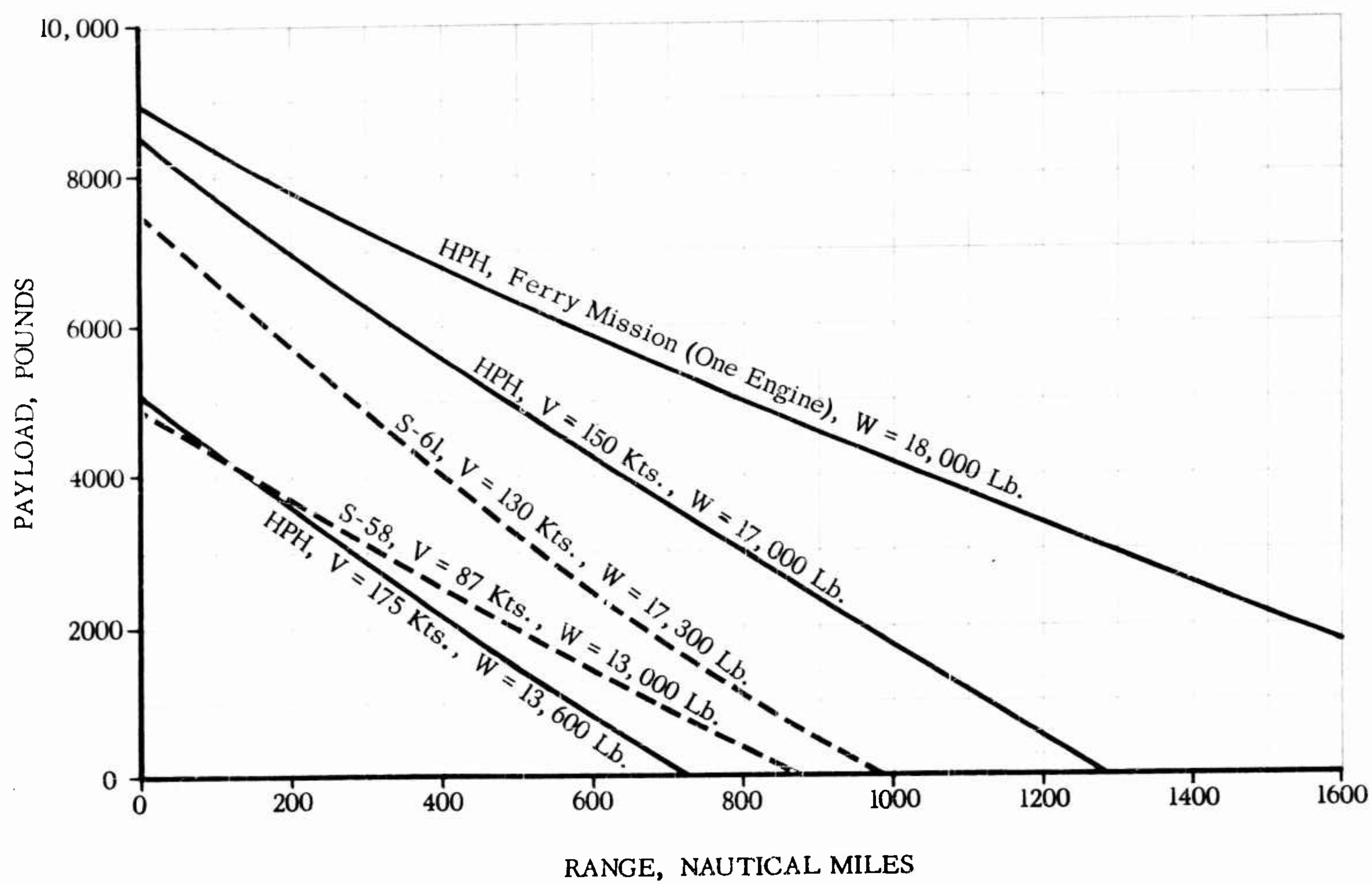


FIGURE 56. PAYLOAD-RANGE CHARACTERISTICS OF THE HIGH PERFORMANCE HELICOPTER

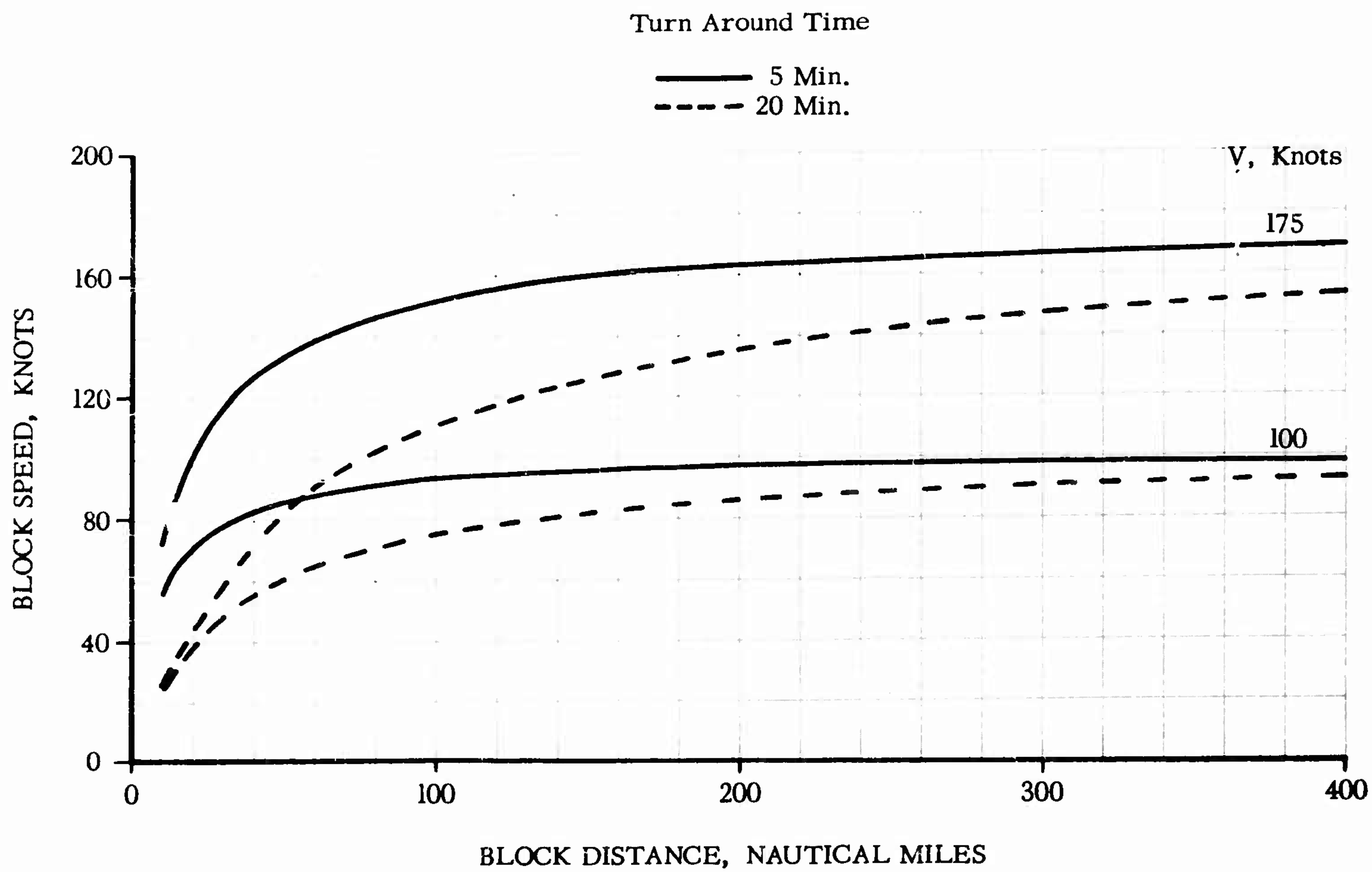


FIGURE 57. EFFECT OF FLIGHT SPEED ON BLOCK SPEED

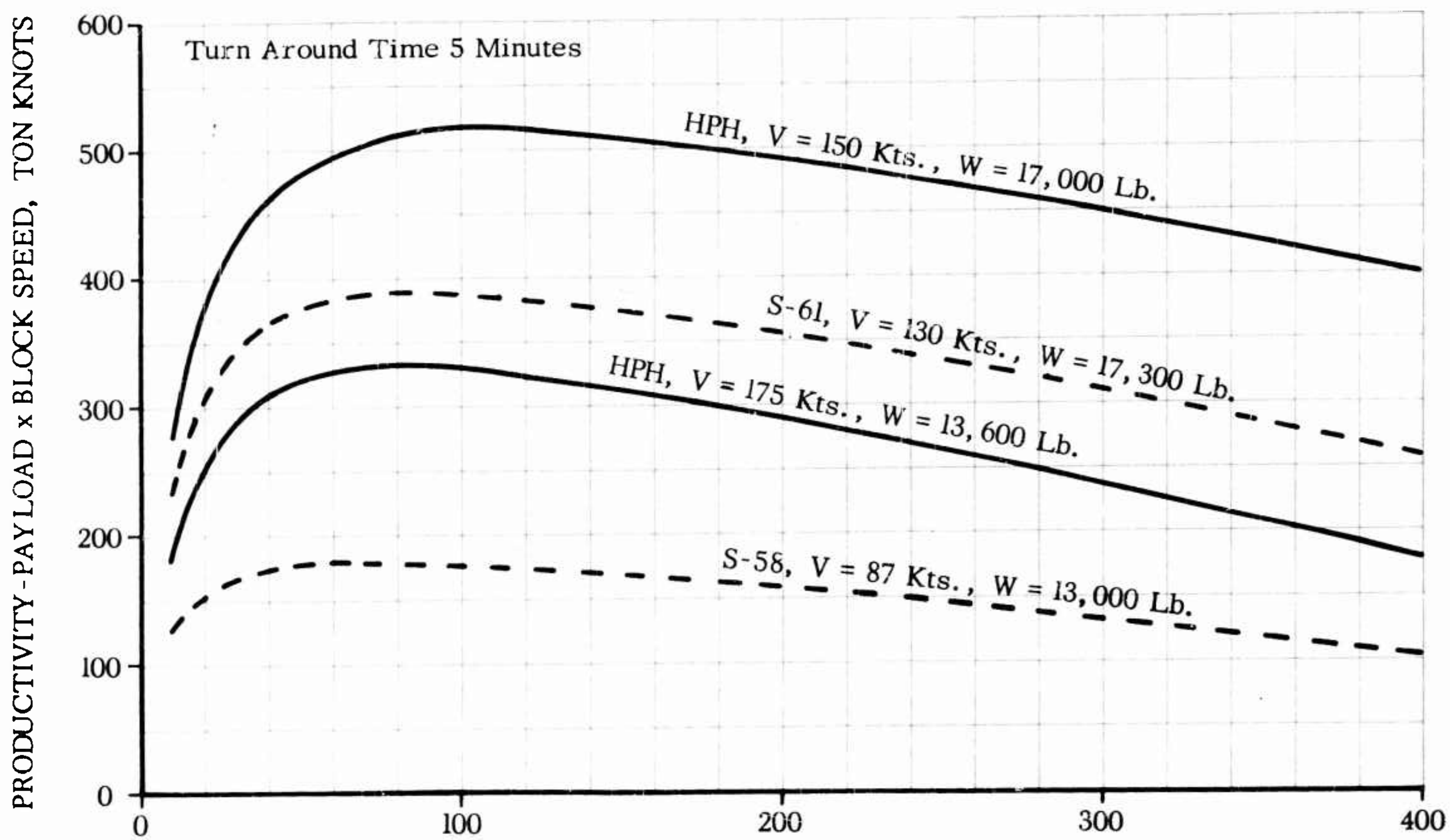


FIGURE 58. PRODUCTIVITY, TURN AROUND TIME 5 MINUTES

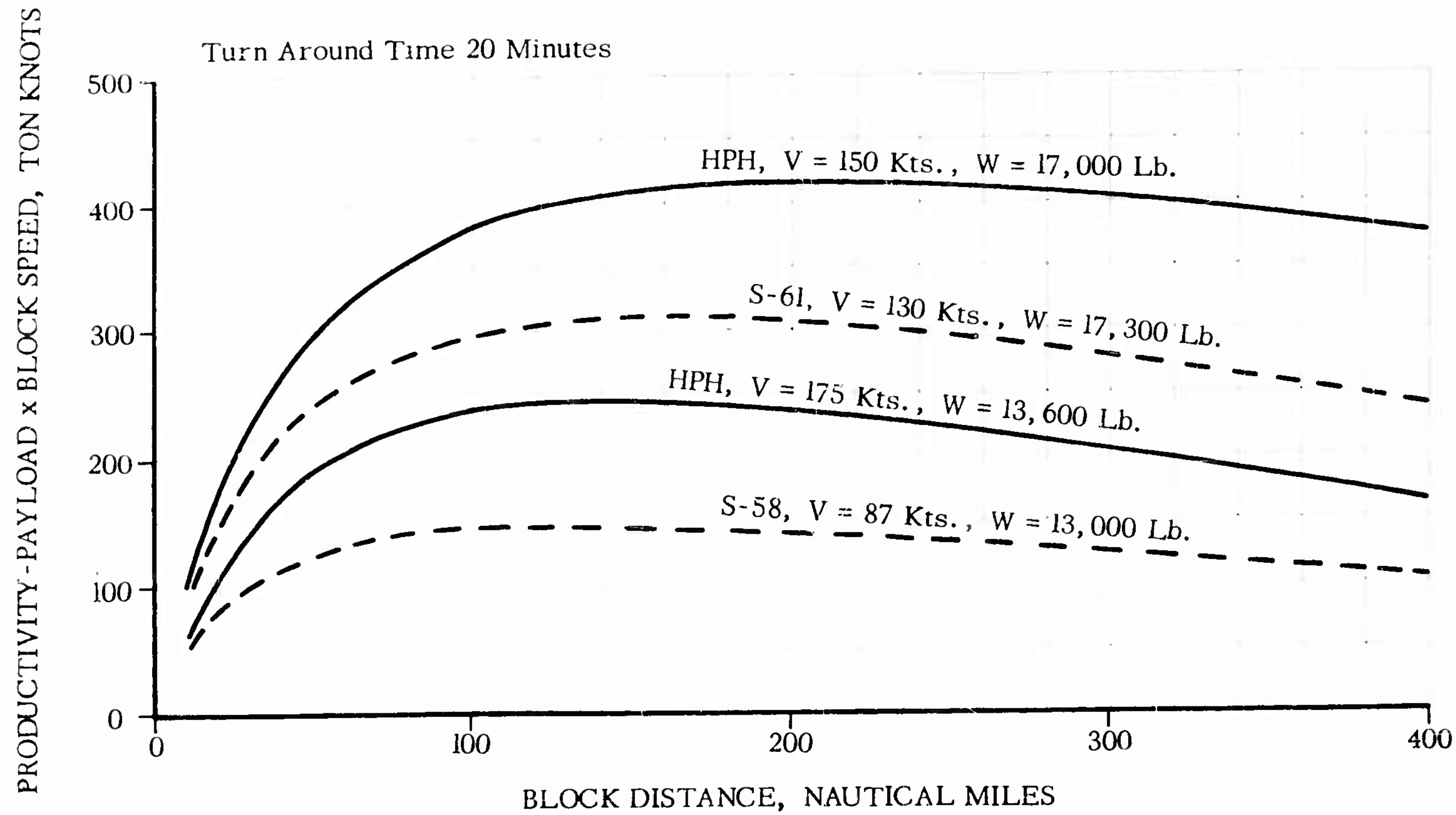


FIGURE 59. PRODUCTIVITY, TURN AROUND TIME 20 MINUTES

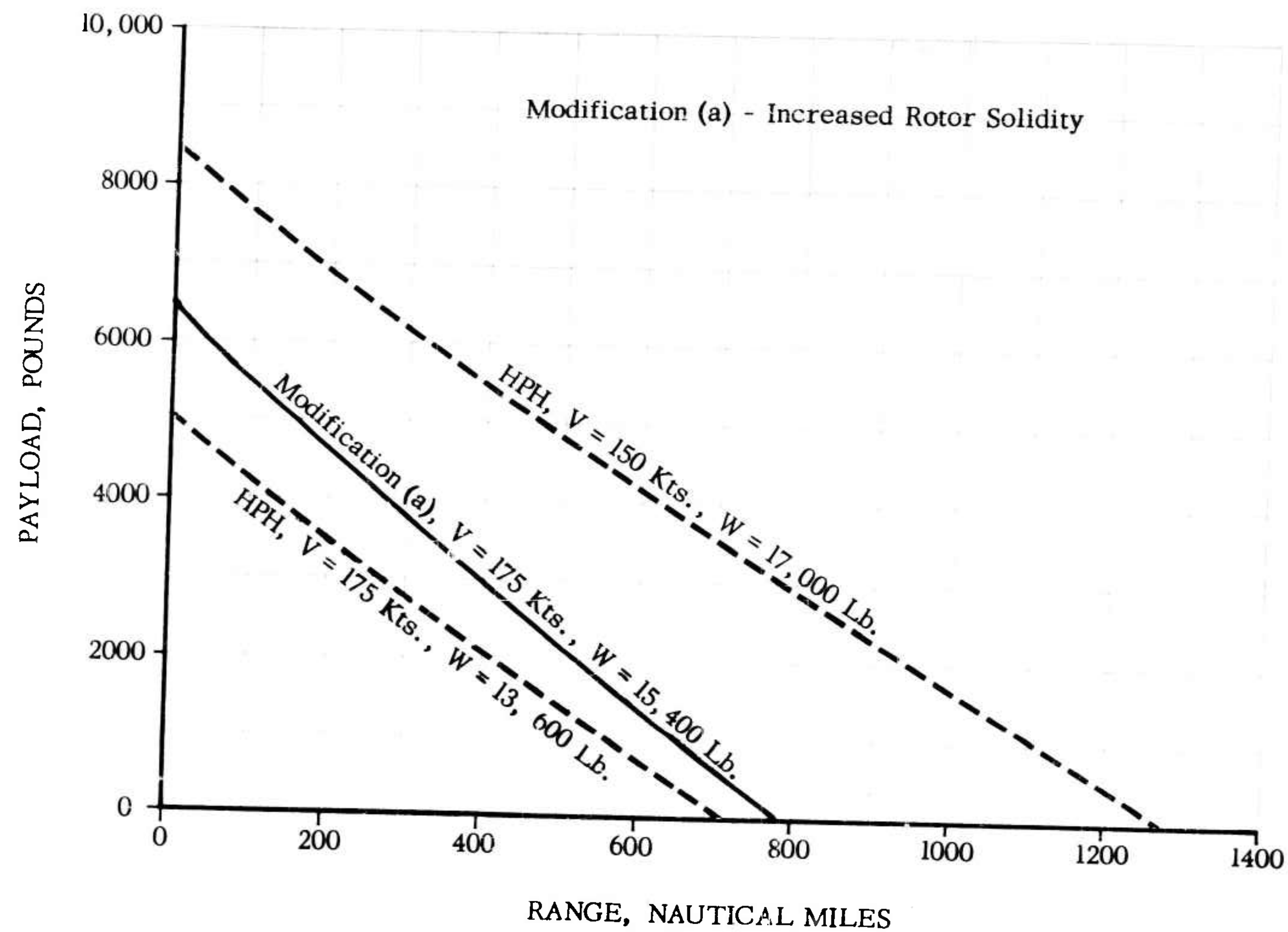


FIGURE 60. PAYLOAD-RANGE CHARACTERISTICS, MODIFICATION (a)

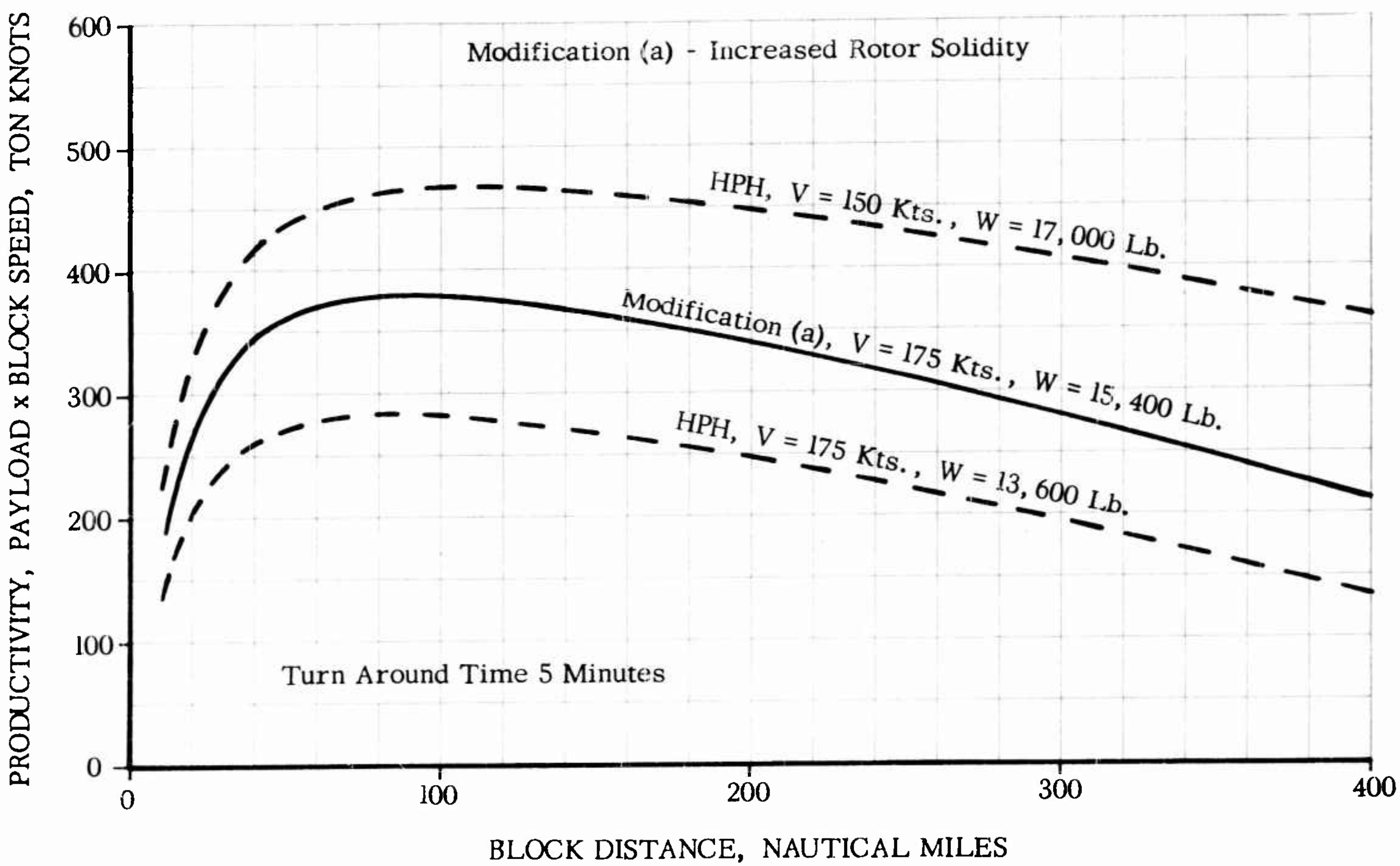
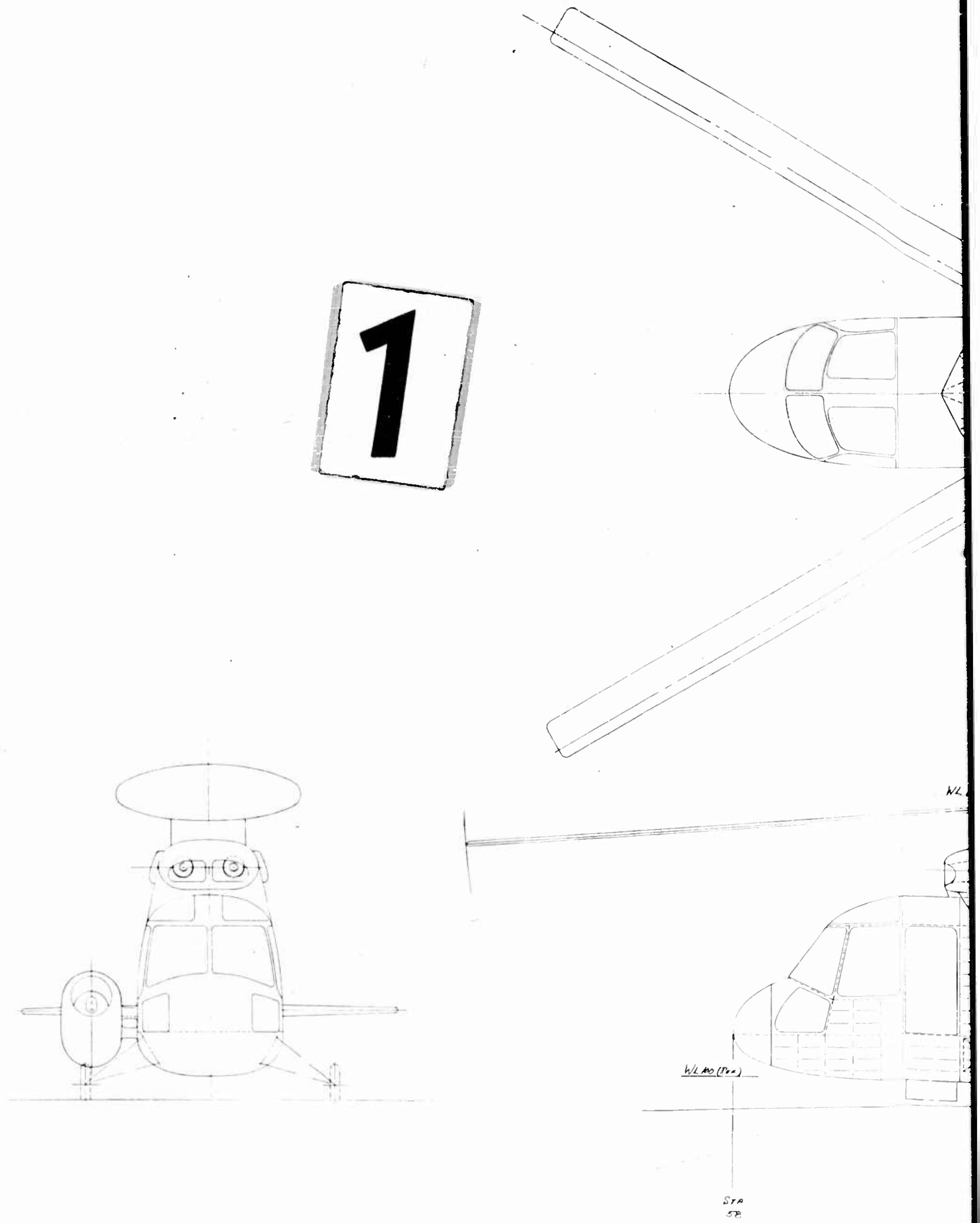
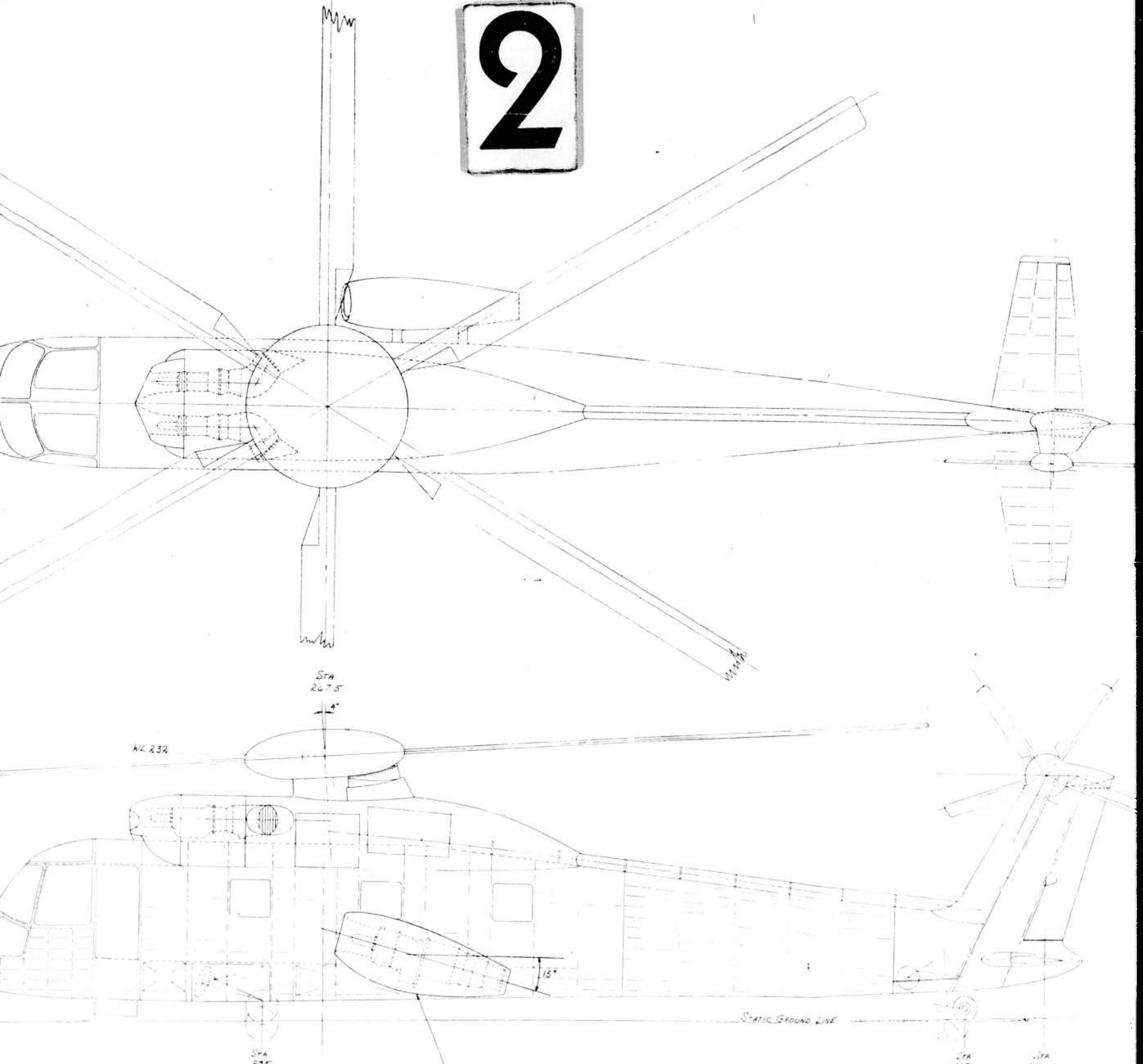


FIGURE 61. PRODUCTIVITY, MODIFICATION (a)

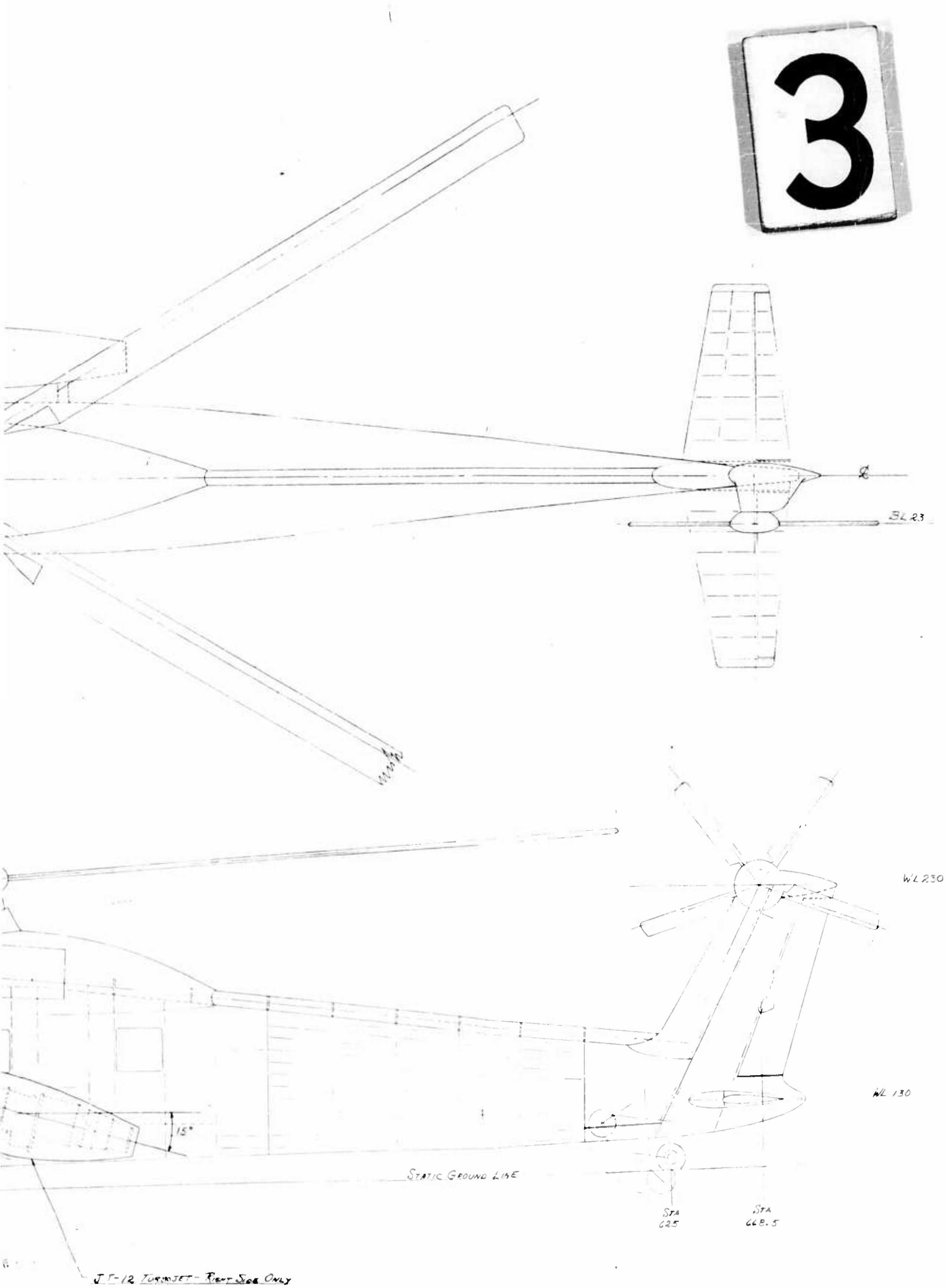
1



2



STA 267.5
4°
WL 232
15°
STATIC GROUND LINE
STA 235
STA
STA 668.5
JT-12 TURBOJET - RIGHT SIDE ONLY



MAIN ROTOR

DIA	56 FT
NO BLADES	6
CHORD	8.25 IN
SOLIDITY	.094
TWIST	-4°

TAIL ROTOR

DIA	10 FT
NO BLADES	5
CHORD	7 11/32 IN
SOLIDITY	.195

HORIZONTAL TAIL

AREA - TOTAL	50 FT²
ELEVATORS	15 FT²
SECTION	NACA 0015
INCIDENCE	0°

FIN AND RUDDER

AREA - TOTAL	40 FT²
RUDDER	10 FT²
SECTION	NACA 0020

ROTOR POWER

2-GE - T58 - 8 ENGINES

AUXILIARY THRUST

P&W - JT-12 TURBOJET

FIG. 62 MOD.(b)
GENERAL ARRANGEMENT
HIGH PERFORMANCE
HELICOPTER WITH JET

SIKORSKY AIRCRAFT
STRATFORD, CONN.
DIVISION OF
UNITED AIRCRAFT CORPORATION

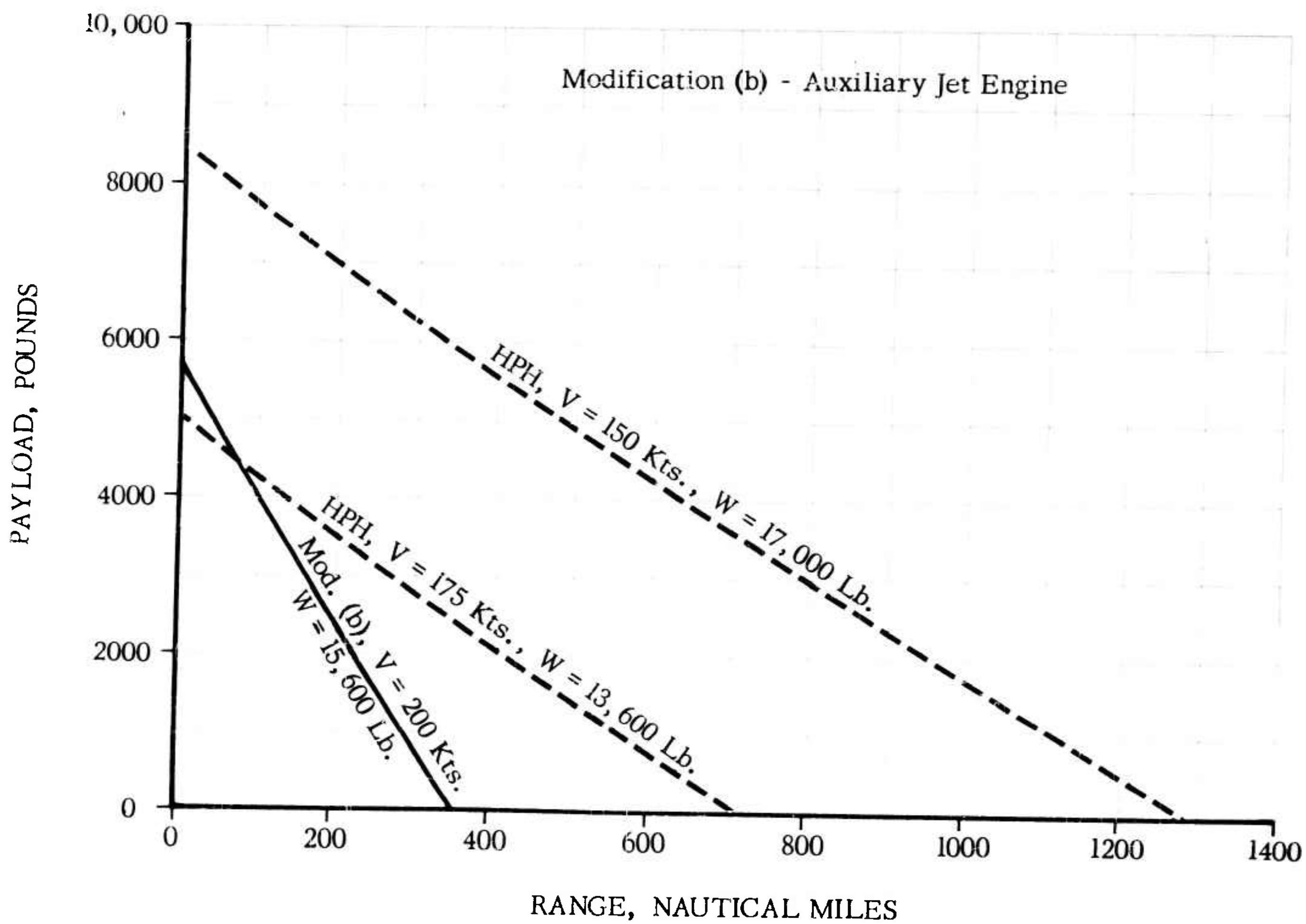


FIGURE 63. PAYLOAD-RANGE CHARACTERISTICS, MODIFICATION (b)

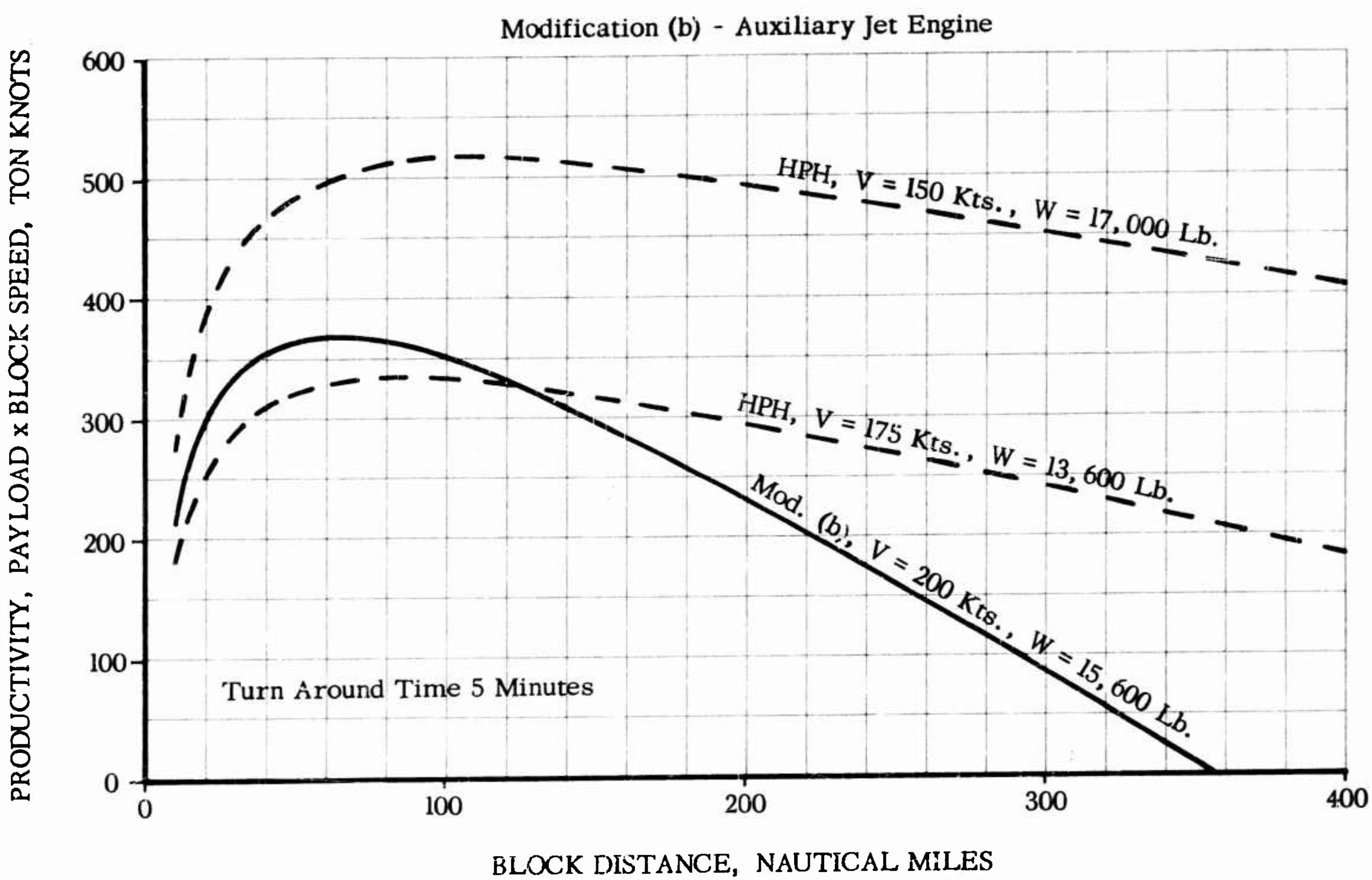
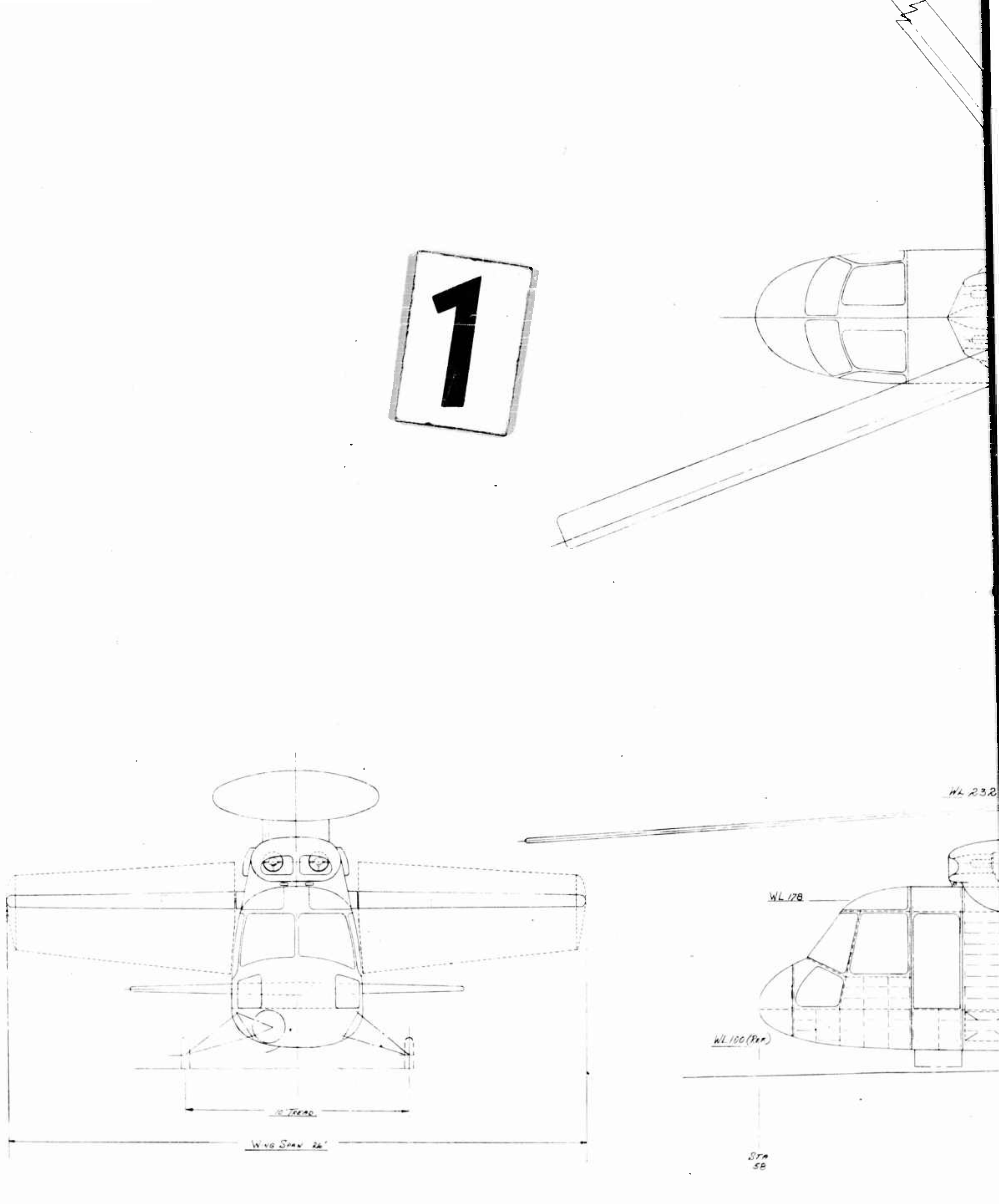
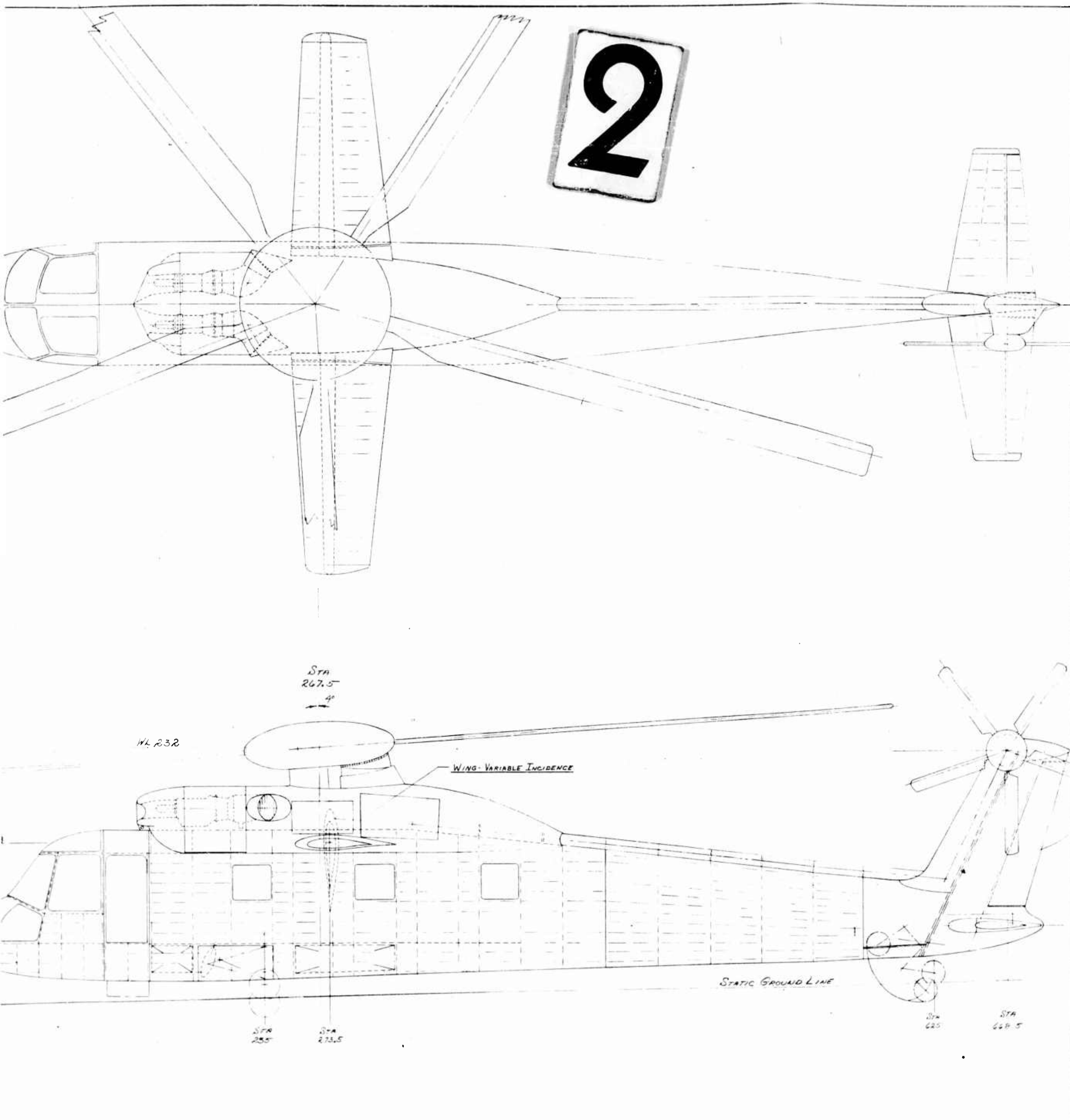
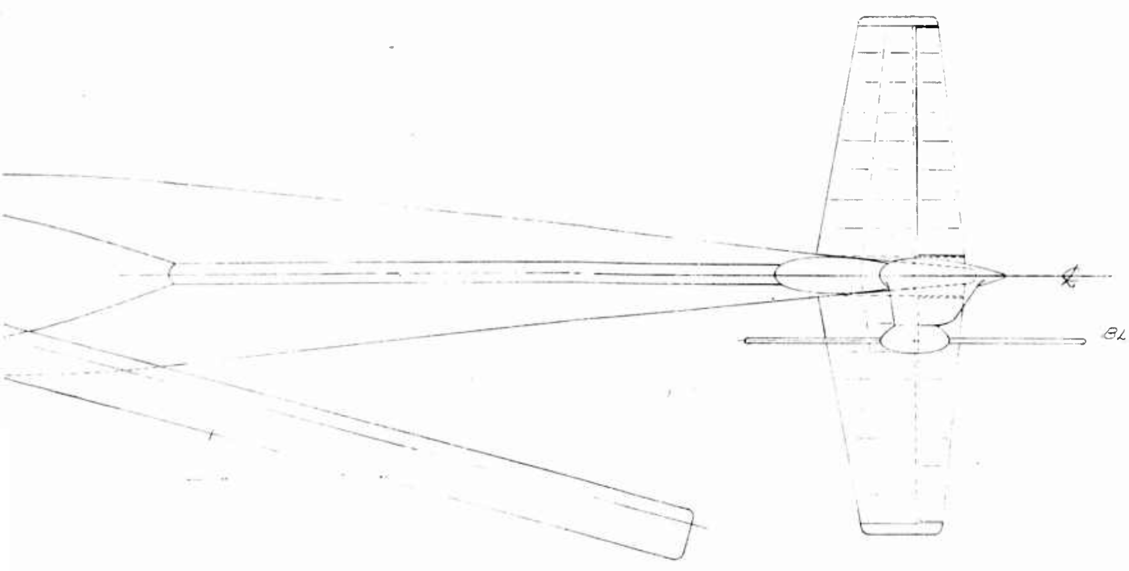


FIGURE 64. PRODUCTIVITY, MODIFICATION (b)

1




2
STA
267.5
4°
WL 232
WING-VARIABLE INCIDENCE
STATIC GROUND LINE
STA
235
STA
273.5
STA
625
STA
668.5



MAIN ROTOR	
DIA.	56 FT
NO BLADES	5
CHORD	18.25 IN
SOLIDITY	.086
TWIST	-9°

TAIL ROTOR	
DIA	10 FT
NO BLADES	5
CHORD	7.39 IN.
SOLIDITY	.195

HORIZONTAL TAIL	
AREA - TOTAL	50 FT^2
ELEVATORS	15 FT^2
SECTION	NACA 0015
INCIDENCE	0°

FIN AND RUDDER	
AREA - TOTAL	40 FT^2
RUDDER	10 FT^2
SECTION	NACA 0020

WING	
AREA	110 FT^2
ASPECT RATIO	6.1
SPAN	26 FT
INCIDENCE	VARIABLE
SECTION	NACA 63_2 A4.5

ROTOR POWER

2 - GE - T58 - 8 ENGINES

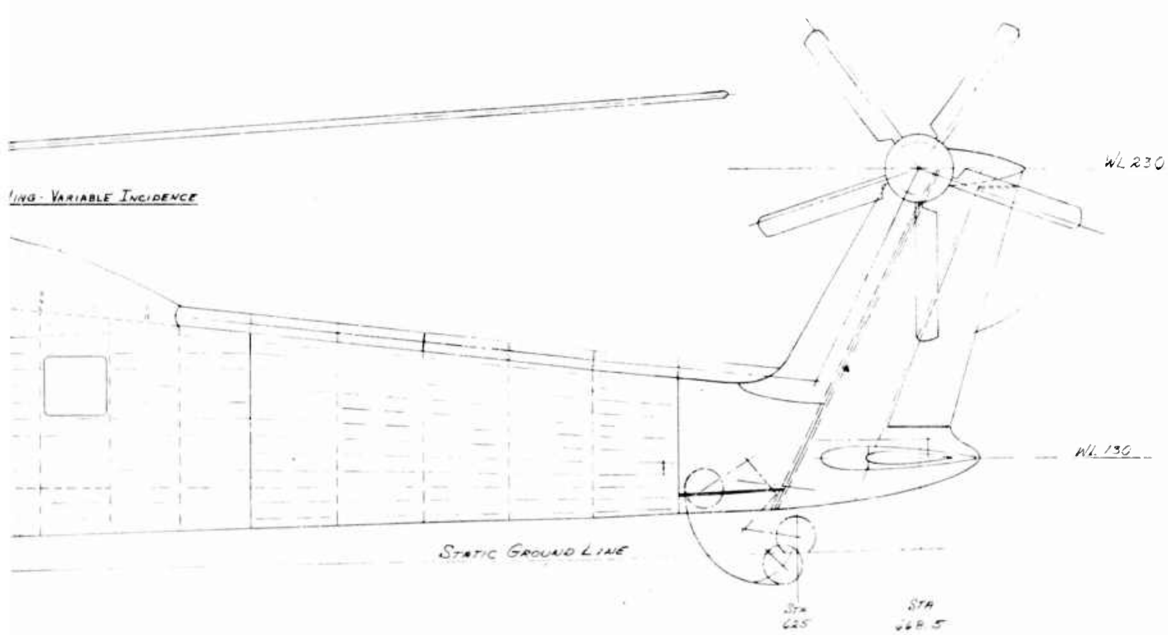


FIG 65 MOD (c)
GENERAL ARRANGEMENT
HIGH PERFORMANCE
HELICOPTER WITH WING

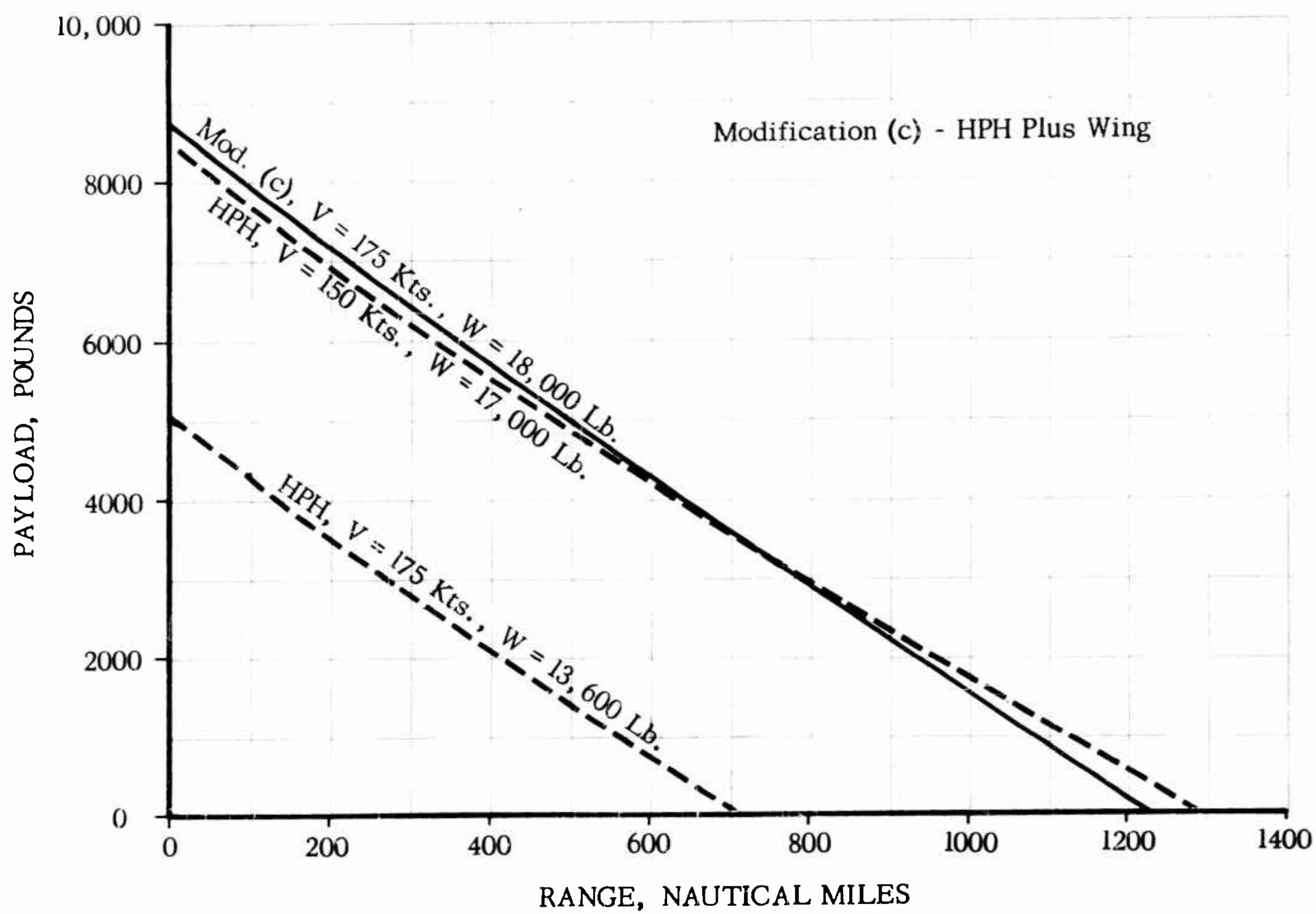


FIGURE 66. PAYLOAD-RANGE CHARACTERISTICS, MODIFICATION (c)

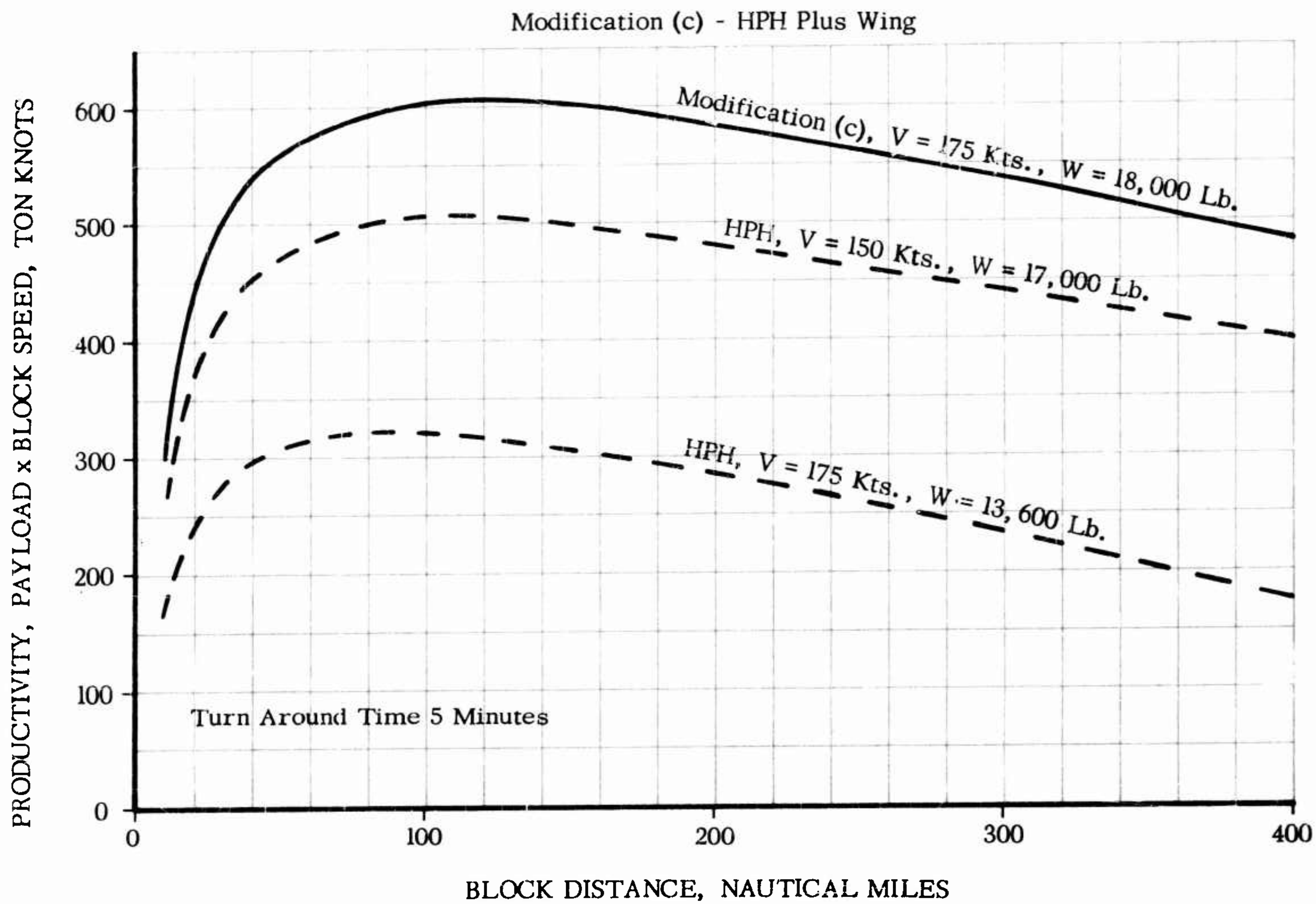
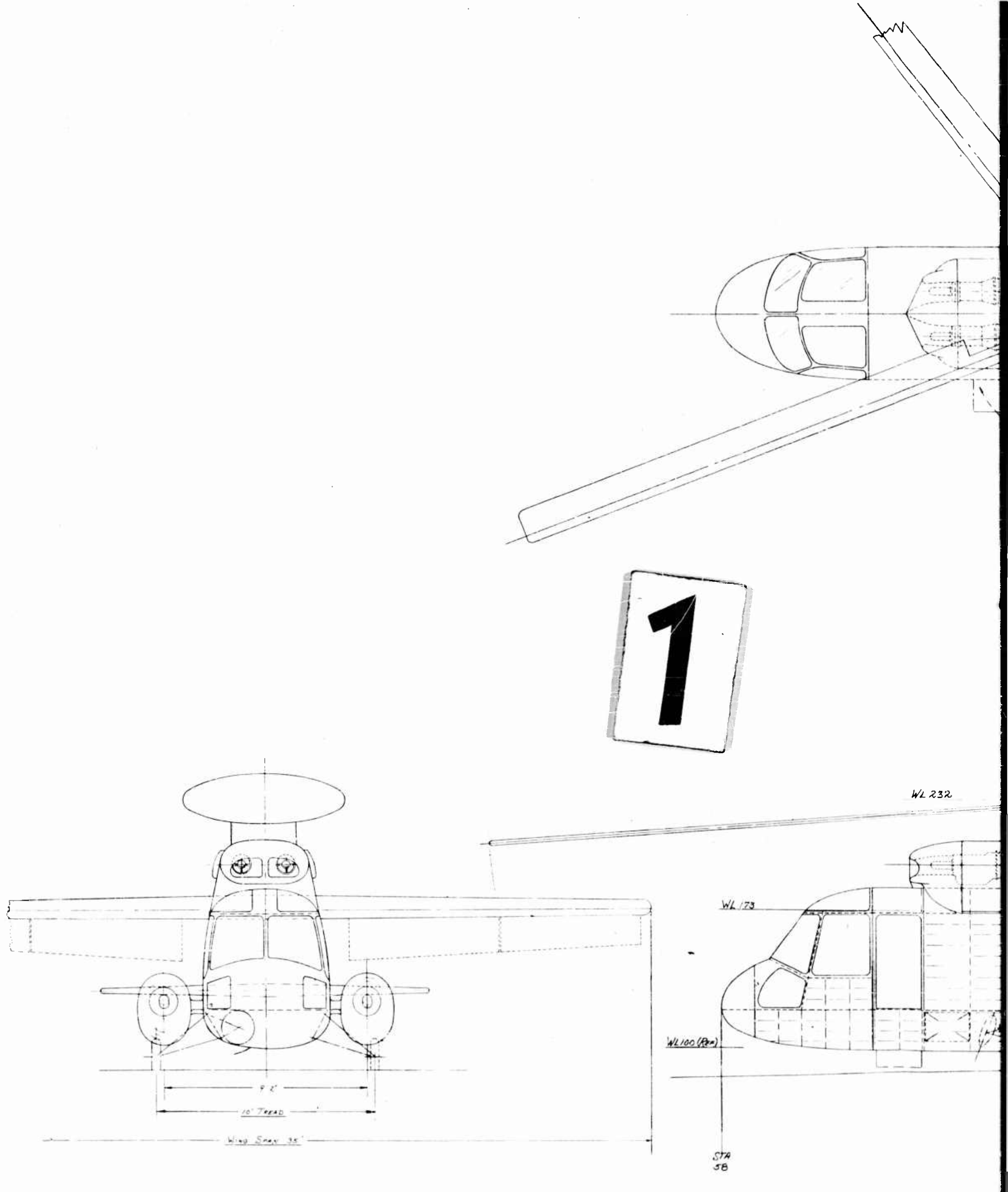
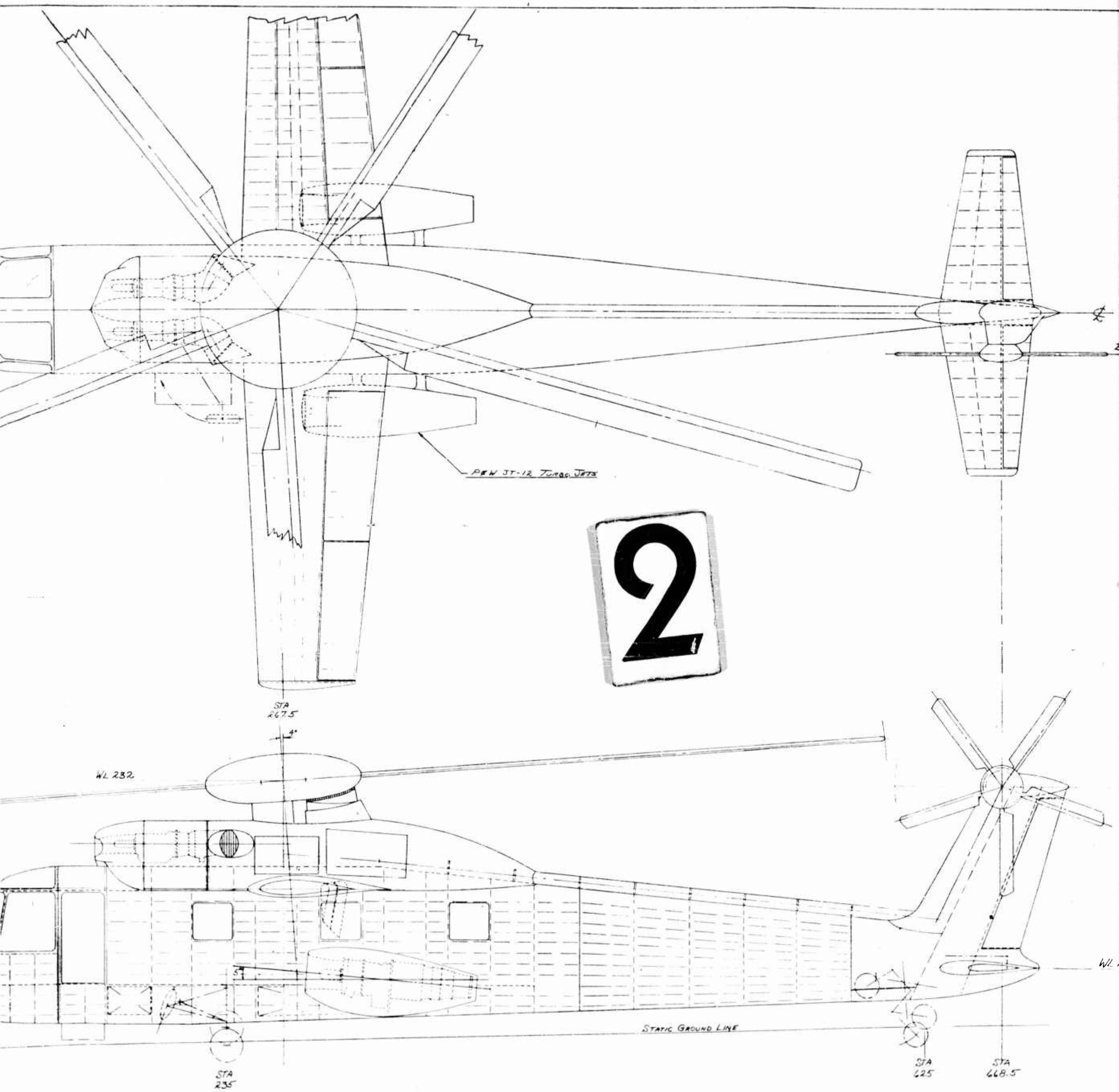


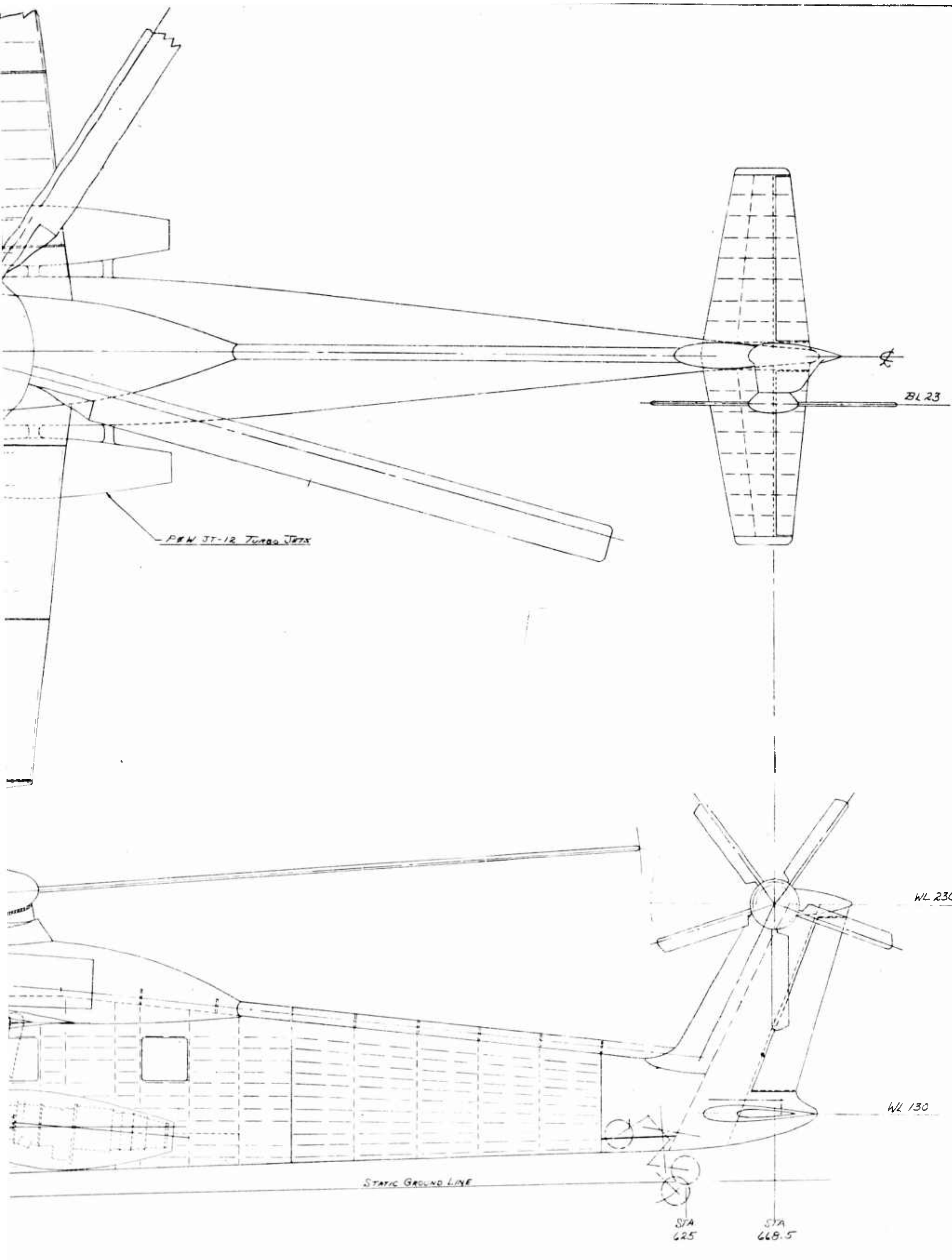
FIGURE 67. PRODUCTIVITY, MODIFICATION (c)



1
WL 232
WL 173
WL100 (REF)
9' 2"
10' TREAD
WING SPAN 35'
STA
58



P&W JT-12 TURBO JETS
2
STA 267.5
4°
WL 232
WL
STATIC GROUND LINE
STA 235
STA 625
STA 668.5



Main Rotor

Dia.	56 FT
No. Blades	5
Chord	18.25 IN.
Solidity	.084
Twist	-4°

Tail Rotor

Dia.	10 FT
No. Blades	5
Chord	7.34 IN
Solidity	.195

Horizontal Tail

Area - Total	50 FT²
- Elevators	15 FT²
Section	NACA 0015
Incidence	0°

Fin and Rudder

Area - Total	40 FT²
Rudder	10 FT²
Section	NACA 0020

Wing

Area - Total	200 FT²
Ailerons & Flaps	57.5 FT²
Aspect Ratio	6.1
Span	35 FT
Section	NACA 63_2A415

Rotor Power

2-GE-T58-8 Engines

Auxiliary Power

2-P&W-JT-12 Engines

3

FIG [illegible]6 MOD. (d)
GENERAL ARRANGEMENT
JET COMPOUND
HELICOPTER

SIKORSKY AIRCRAFT
STRATFORD, CONN.

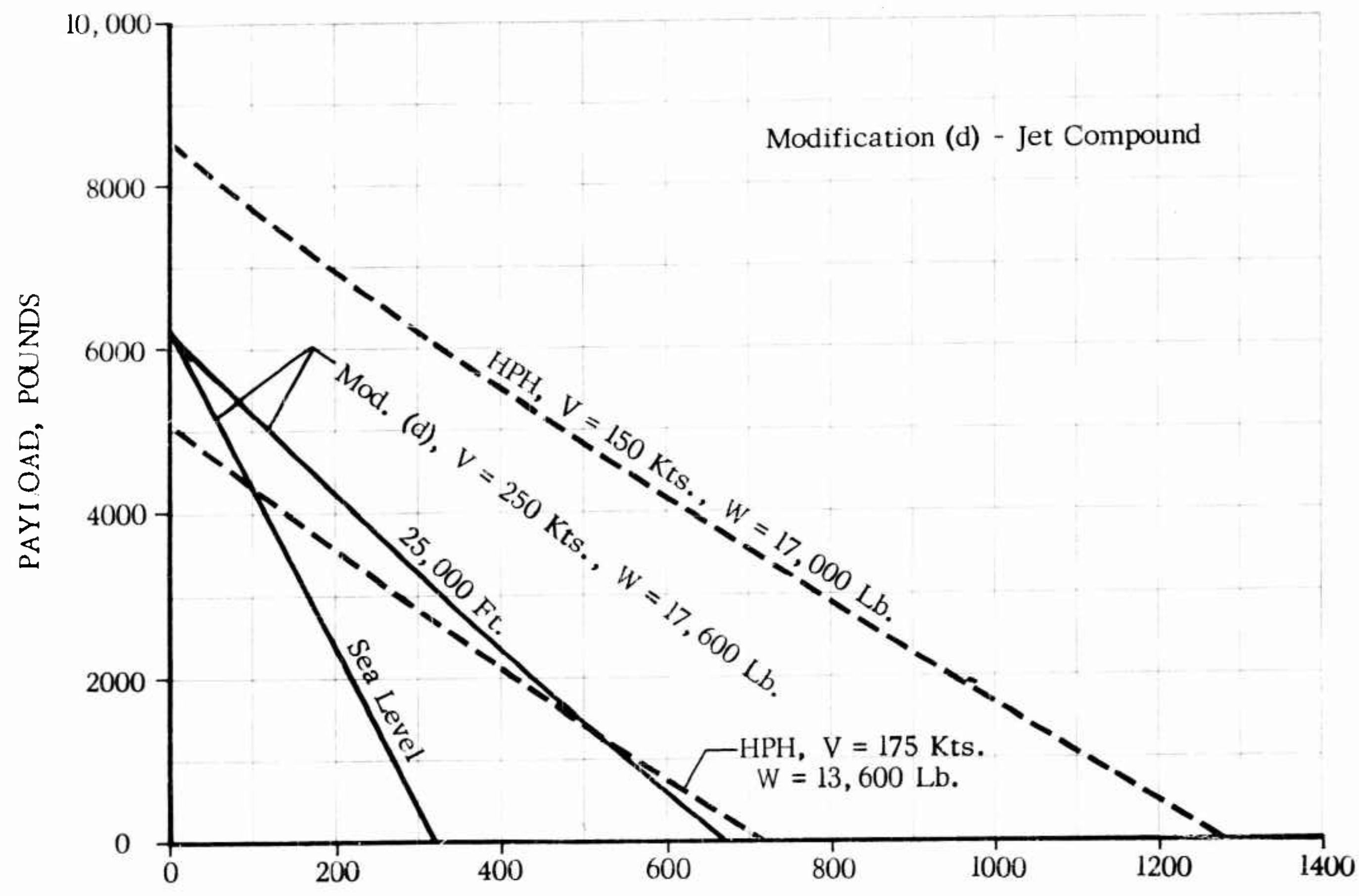


FIGURE 69. PAYLOAD-RANGE CHARACTERISTICS, MODIFICATION (d)

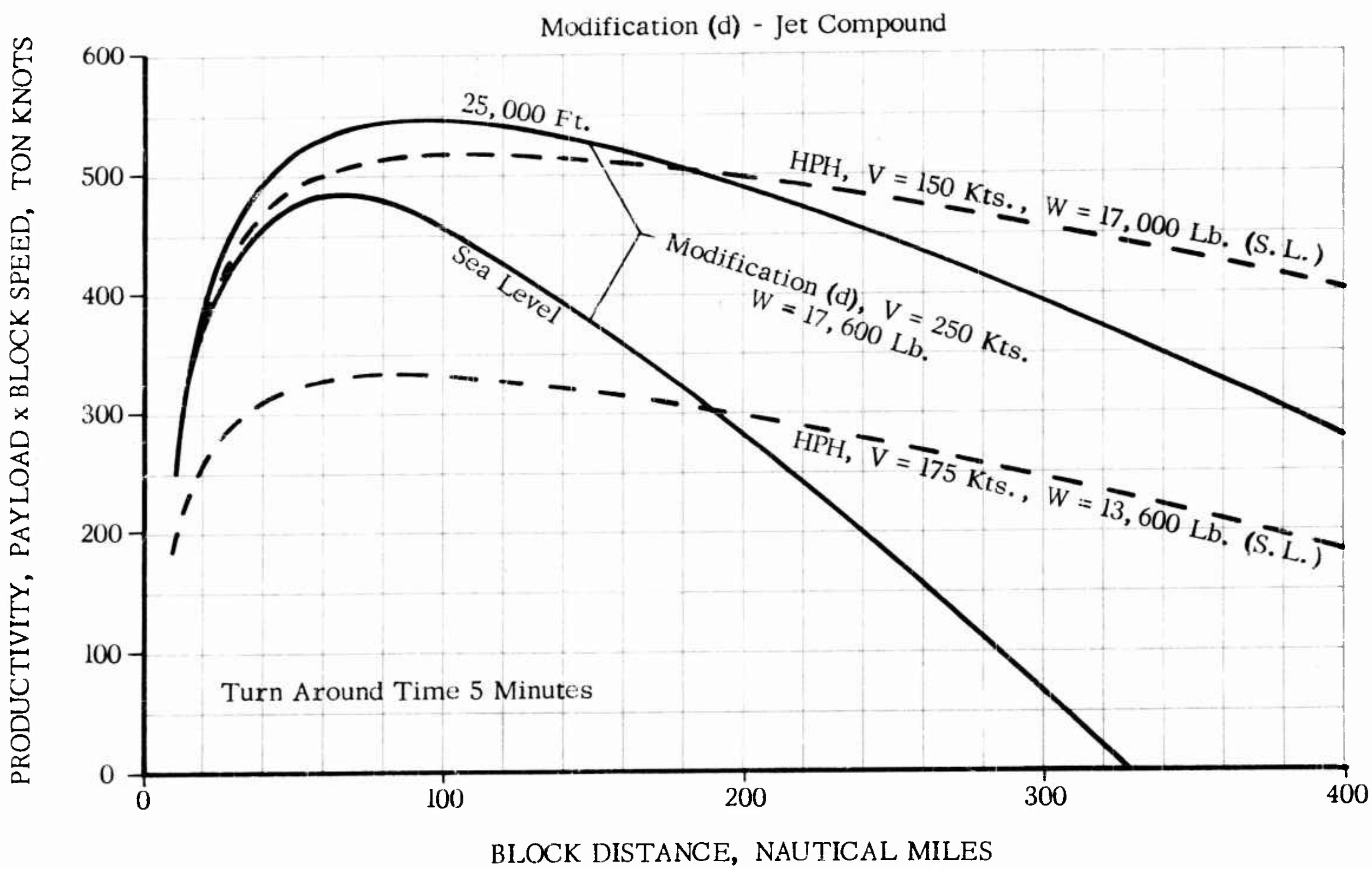


FIGURE 70. PRODUCTIVITY, MODIFICATION (d)

TABLE 2

PHYSICAL CHARACTERISTICS USED IN SIMULATED STUDY

Main Rotor	
diameter	56 ft.
number of blades	5
chord	1.367 ft.
aerodynamic twist	-6 deg.
solidity	.0777
Lock number	10.14
offset of flapping hinge	1 ft.
shaft tilt	3 deg. fwd.
nominal tip speed	650 ft/sec.

Fuselage	Water Line	Station Line
rotor head location	128.4	137.7
tail rotor location	107.2	535.0
c.g. locations forward	44.0	130.0
normal	44.0	137.7
aft	44.0	145.0
horizontal tail	59.0	477.2
vertical tail	66.5	513.0
moments of inertia roll	5190 slug ft.^{2}	
pitch	25,500 slug ft.^{2}	
yaw	21,600 slug ft.^{2}	
minimum parasite area	13 ft.^{2}	

Tail Rotor	
diameter	9.33 ft.
number of blades	4
chord	.612 ft.
aerodynamic twist	0
solidity	.166
Lock number	4.96
nominal tip speed	650 ft/sec.

Horizontal Tail	
area	50 ft.^{2}
aspect ratio	4
airfoil section	NACA 0015
incidence	0 deg.

NOTE: Not identical to final design values

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AD ______________ Accession No. ______________

Sikorsky Aircraft Division, United Aircraft Corporation, Stratford, Connecticut. HIGH PERFORMANCE SINGLE ROTOR HELICOPTER STUDY - Evan A. Fradenburgh

Report No. TREC 61-44, April, 1961, 147 pp. (Contract DA 44-177-TC-648) USATRECOM Proj 9R38-13-014-01, Unclassified Report

A preliminary design study was conducted by Sikorsky Aircraft to demonstrate the benefits of a high performance helicopter configuration. The design cruise speed was 175 knots with a payload of two tons, and the minimum ferry range was set at 1600 nautical miles plus a one hour fuel reserve.

The design was based on the powerplant and rotor system of the Sikorsky S-61 helicopter, and could be built and flown in the immediate future. A large reduction in parasite drag compared to current helicopters was achieved, and the minimum performance objectives were met or exceeded. Flying qualities satisfied MIL SPEC 8501 without artificial stabilization.

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II Contract DA 44-177-TC-648

AD ______________ Accession No. ______________

Sikorsky Aircraft Division, United Aircraft Corporation, Stratford, Connecticut. HIGH PERFORMANCE SINGLE ROTOR HELICOPTER STUDY - Evan A. Fradenburgh

Report No. TREC 61-44, April, 1961, 147 pp. (Contract DA 44-177-TC-648) USATRECOM Proj 9R38-13-014-01, Unclassified Report

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